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FINAL REPORT

VOYAGER SPACECRAFT

PHASE B, TASK D

VOLUME IV (BOOK 2 OF 5)

APPLICABILITY OF APOLLO CHECKOUT EQUIPMENT

PREPARED FOR

GEORGE C. MARSHALL SPACE FLIGHT CENTER

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VOLUME SUMMARY

The Voyager Phase B, Task D Final Report is contained in four volumes. The volume numbers and titles are as follows:

Volume I	Summary
Volume II	System Description
Book 1	Guidelines and Study Approach, System Functional Description
Book 2	Telecommunication
Book 3	Guidance and Control Computer and Sequencer Power Subsystem Electrical System
Book 4	Engineering Mechanics Propulsion Planet Scan Platform
Book 5	Design Standards Operational Support Equipment Mission Dependent Equipment
Volume III	Implementation Plan
Volume IV	Engineering Tasks
Book 1	Effect of Capsule RTG's on Spacecraft
Book 2	Applicability of Apollo Checkout Equipment
Book 3	Central Computer
Book 4	Mars Atmosphere Definition
Book 5	Photo-Imaging

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SECTION 1

INTRODUCTION

1.1 OBJECTIVE

The basic objective of this engineering task was to determine the applicability to the Voyager program of the checkout system and support equipment now in use at the Kennedy Space Center (KSC) on the Saturn/Apollo program. The specific objectives were:

a. The determination of the extent of applicability of the checkout systems at KSC to Voyager system test and launch control, and the nature of any incompatibilities.

b. The determination of the extent and nature of the engineering modifications required to achieve compatibility with Voyager.

c. The identification of ground service hardware which could either perform or be modified to perform analogous functions for Voyager.

1.2 SCOPE

The Saturn/Apollo equipment considered during the study included: Apollo's Acceptance Checkout Equipment (ACE-S/C); the Saturn Electrical Support Equipment (ESE); and the Apollo and Saturn Mechanical and Fluid Ground Equipment.

The capabilities of this equipment were analyzed and compared to Voyager requirements. Where modifications were indicated, their extent (cost and feasibility) were established.

The General Electric Voyager Task B Study and preferred design spacecraft, updated with Task D designs, test requirements, and flow plans where appropriate, was utilized as the baseline for this Study.

1.3 REPORTS

This document summarizes and extends the studies included in the following Milestone Reports issued during Task D:

a. Preliminary Voyager/ACE System Description, VOY-D2-TM1, dated 19 July 1967.

b. Voyager ACE Trade Studies, VOY-D2-TM15, dated 18 August 1967.

c. Applicability of Saturn/Apollo Mechanical and Fluid Ground Equipment to Voyager, VOY-D2-TM21, dated 11 September 1967.

1.4 CONTENTS

Section 2 of this report contains a summary of the significant results and recommendations regarding the applicability of Apollo/Saturn OSE (Operational Support Equipment) to Voyager.

Section 3 indicates, primarily by flow diagrams, the requirements for Voyager OSE.

Section 4, after briefly describing the Apollo ACE, discusses its applicability to Voyager and indicates the required and desired modifications.

Section 5 discusses the rationale utilized in screening Saturn/Apollo mechanical and fluid ground equipment for Voyager and presents recommended applicable equipment items.

Section 6 indicates the key reference documents utilized in the study.

In addition, the following Appendixes have been included to amplify certain aspects of the study:

Appendix A is a summary description of Apollo ACE.

Appendix B presents a description of the Voyager SIE (spacecraft interface equipment).

Appendix C discusses Voyager launch equipment requirements.

Appendix D consists of data sheets describing the recommended Voyager - applicable Apollo/Saturn mechanical and fluid ground equipment.

SECTION 2

SUMMARY AND CONCLUSIONS

2.1 CHECKOUT SYSTEMS

Two checkout systems are operational at the KSC Saturn/Apollo facilities: the Saturn ESE (electrical support equipment) and the Apollo ACE-S/C (acceptance checkout equipment - spacecraft). Of these, only ACE has a significant degree of applicability. This is attributable to the fact that as the ESE system is sized for the Saturn application, it is more comprehensive than needed for Voyager; moreover, its use is assumed to be pre-empted by the Saturn V launch vehicle used for the Voyager mission and for any other in-process Apollo-type missions.

Most of ACE (four systems of which are currently operational at KSC) is located in the Manned Space Operations Building (MSOB) at Merritt Island; some elements of the system, however, stay with Apollo spacecraft through its ground mission flow at KSC, as shown in the following top level block diagram (Figure 2-1).

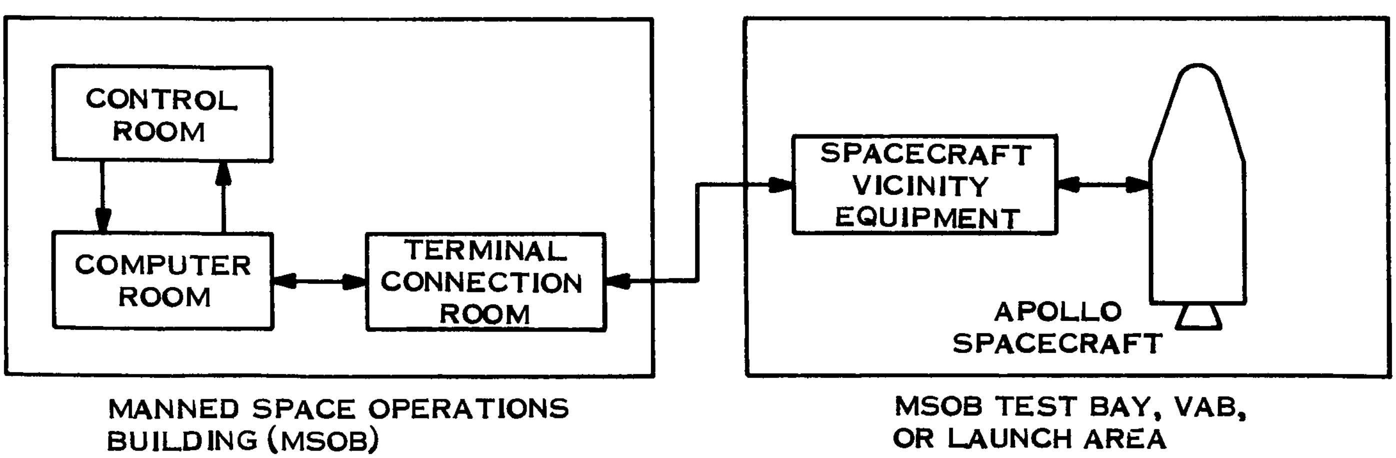

Figure 2-1. Location of ACE at KSC

The heart of the system--the computers, controls and displays located in the MSOB--was generally found to be applicable to Voyager checkout; however, for the most part, the spacecraft vicinity equipment must be unique to Voyager.

More specifically, operators' controls and displays, located in the MSOB control room, can be adapted to Voyager on a one-for-one (one ACE for one Voyager spacecraft) basis. Most of the required control and display reconfiguration can be accomplished through test programming, as these controls and displays are general-purpose in nature. An optional modification would involve relocation of switches, lights, and meters on the panels to facilitate test operations.

The ACE computer room, which contains a complex of two CDC 160G computers, two Radiation 540 decommutators, and other peripheral equipment, is apparently applicable, in its existing form, to Voyager, with modification to accommodate potential Voyager decommutation incompatibility and trend testing. The preferred approach, in the event Voyager spacecraft telemetry would use asynchronous medium- and low-speed telemetry decks, would be to have ACE's decommutator handle OSE telemetry while the downlink computer handles the spacecraft telemetry decommutation. Various approaches were established to implement 10-bit accuracy (which would allow for more sensitive trend detection) with minimum impact on the present eight-bit system.

The terminal connection room of the MSOB does not appear to need any modification to be applied to Voyager.

The present ACE system software packages exist in two parts: an essentially system oriented package for computer implementation of operator generated commands, data analysis and display functions, and an adaptive program for automated testing. The majority of the former package is applicable to Voyager; the adaptive test programs, however, are written for specific spacecraft and the Apollo routines will not be suitable for Voyager.

Although ACE spacecraft vicinity equipment is less susceptible to "universal" use than control room and computer room equipment, there is the possibility of utilizing certain Saturn/Apollo components in implementing Voyager vicinity equipment. Examples of these are the Saturn DDAS (Digital Data Acquisition System) as a 10-bit PCM equipment and the Apollo baseplate units as standard switching modules.

2.2 SUPPORT EQUIPMENT

Approximately 100 items of Apollo fuel, oxydizer and gas service equipment, and handling equipment have been identified as capable, with various degrees of modification, of being adapted to Voyager. Most of these items are fluid and fuel service equipment used for LEMDE. Additionally, due to location, size, and weight similarity between LEM and Voyager, many items of LEM handling equipment can be utilized with a minimum of modification.

2.3 CONCLUSIONS AND RECOMMENDATIONS

This study has established the technical feasibility of applying significant portions of ACE-S/C to Voyager checkout and launch control. (Section 5 of Volume II evaluates alternative OSE concepts from an operational and implementation point of view.)

A program decision, however, will additionally require consideration of:

a. Program priorities for use of the ACE systems at KSC.

b. Overall program costs to make required modifications and to provide the necessary sets of equipment at the factory as well as at KSC.

c. Projected performance of ACE in 1973, 1975, etc., versus development of a state-of-the-art and Voyager-optimized system.

Insofar as LEM fluid service and handling equipment utilization is concerned, technical feasibility (with a minimum of modification for many of the selected items) appears established. Decisions for utilization would, therefore, be primarily dependent upon the availability of this equipment for Voyager.

SECTION 3
VOYAGER PROGRAM REQUIREMENTS

The Voyager OSE must be capable of supporting the test and operation of the proof test model and the flight spacecraft during factory acceptance tests and the KSC re-acceptance test and launch operation. The spacecraft flow during these operations is shown in Figures 3-1, 3-2, and 3-3 respectively.

In-depth testing is performed during the majority of these operations in order to establish trends of operating characteristics as a function of time (age), temperature, and voltage. Analysis of this data will then permit identification of the quality of each spacecraft subsystem, so that the commitment of two (of three) flight spacecraft may be confidently accomplished.

Once these tests are complete (i.e., when the spacecraft leaves the spacecraft checkout facility), the prime requirement for the OSE becomes one of monitoring the gross status of the spacecraft under static ambient conditions. This is generally limited to performing simple end-to-end tests or just monitoring the parameters via the spacecraft telemetry. (This includes operations at the launch pad.)

The relative complexity of the system testing may be illustrated by noting that the system tests not only utilize the 35 umbilical functions and the spacecraft command and telemetry links (required to perform the simple end-to-end tests and monitoring functions), but further require 300 spacecraft hardwire functions.

Analysis of the spacecraft flow and quantities (as discussed in the Phase D Implementation Plan) indicates that from five to seven sets of OSE are required to support the Voyager Program. Three of these sets may be time-shared between the factory and KSC if the MSOB units are available and if this proves to be the most effective approach from a cost viewpoint.

Appendix B, Spacecraft Interface Equipment, and Appendix C, Launch Equipment Requirements, outline in further detail the types of tests and capability that are required... and how this capability can be achieved in the OSE for a Voyager-type spacecraft.

TEST AREAS								
	SPACECRAFT ASSEMBLY AND CHECKOUT AREA	Spacecraft System Assembly	Spacecraft System Tests	EMI Tests	Free Mode Tests	Weight & CG Tests	Capsule Mating	Planetary Veh System Tests
		■Integration Tests ■S/C - OSE Compat. Tests	■Functional Capability ■Margin Evaluation ■Signal & Control ■Mission Sequences ■Back-up Modes Tests ■Articulation Tests	■"Conducted" Transients ■RF Environment ■Operating Margins	■Electrically Isolated S/C ■Mission Sequence ■Load Sharing	Alignment Checks	■Mechanical Compatibility ■Elec. Compatibility -Power -Signal ■EMI Checks	■Mission Seque ■All Modes Tes ■Parameter Var
	SOLAR SIMULATOR							
	ACOUSTIC TEST FACILITY							
	STATIC FIRING TEST FACILITY							

FOLDOUT FRAME

3-2A

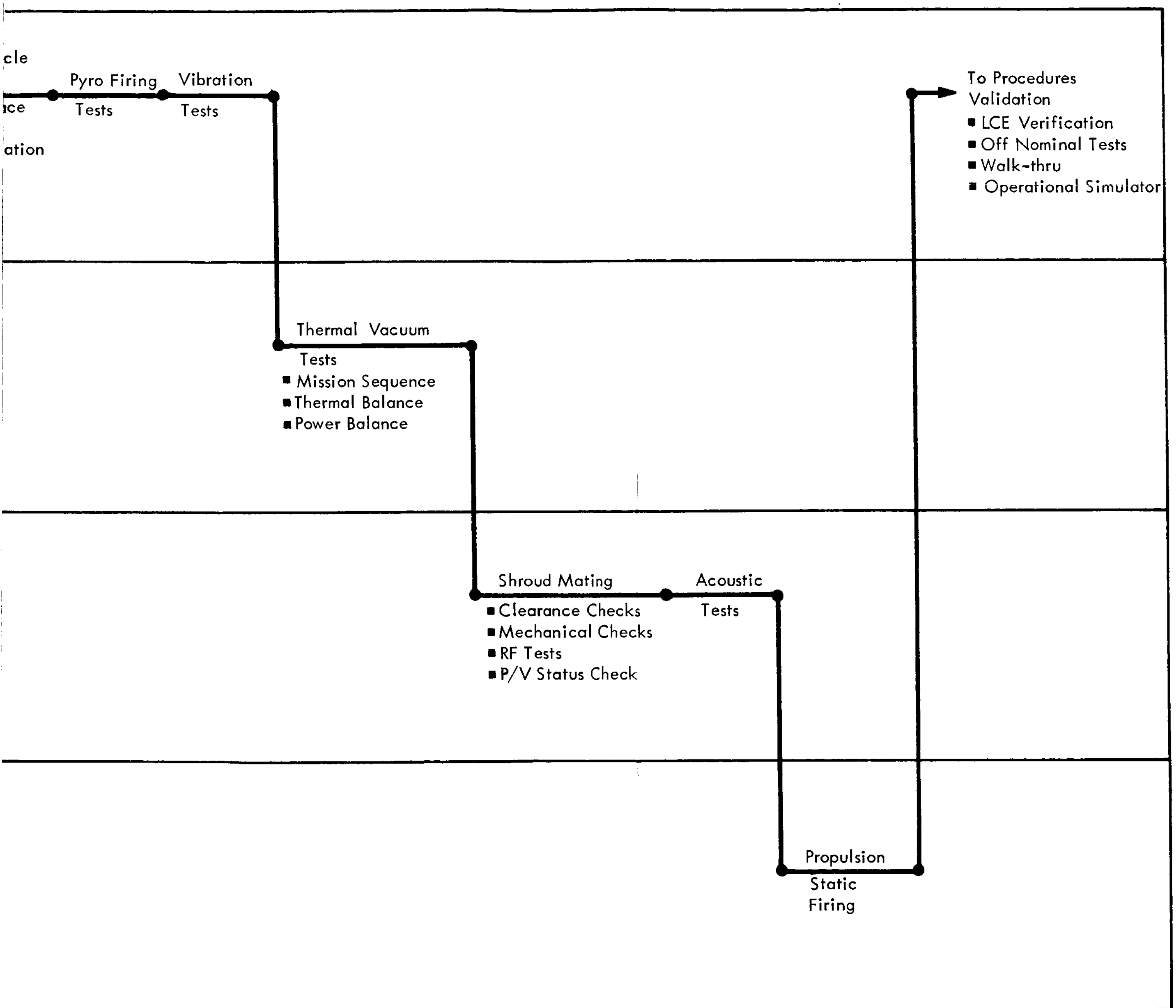

Figure 3-1. Proof Model Flow

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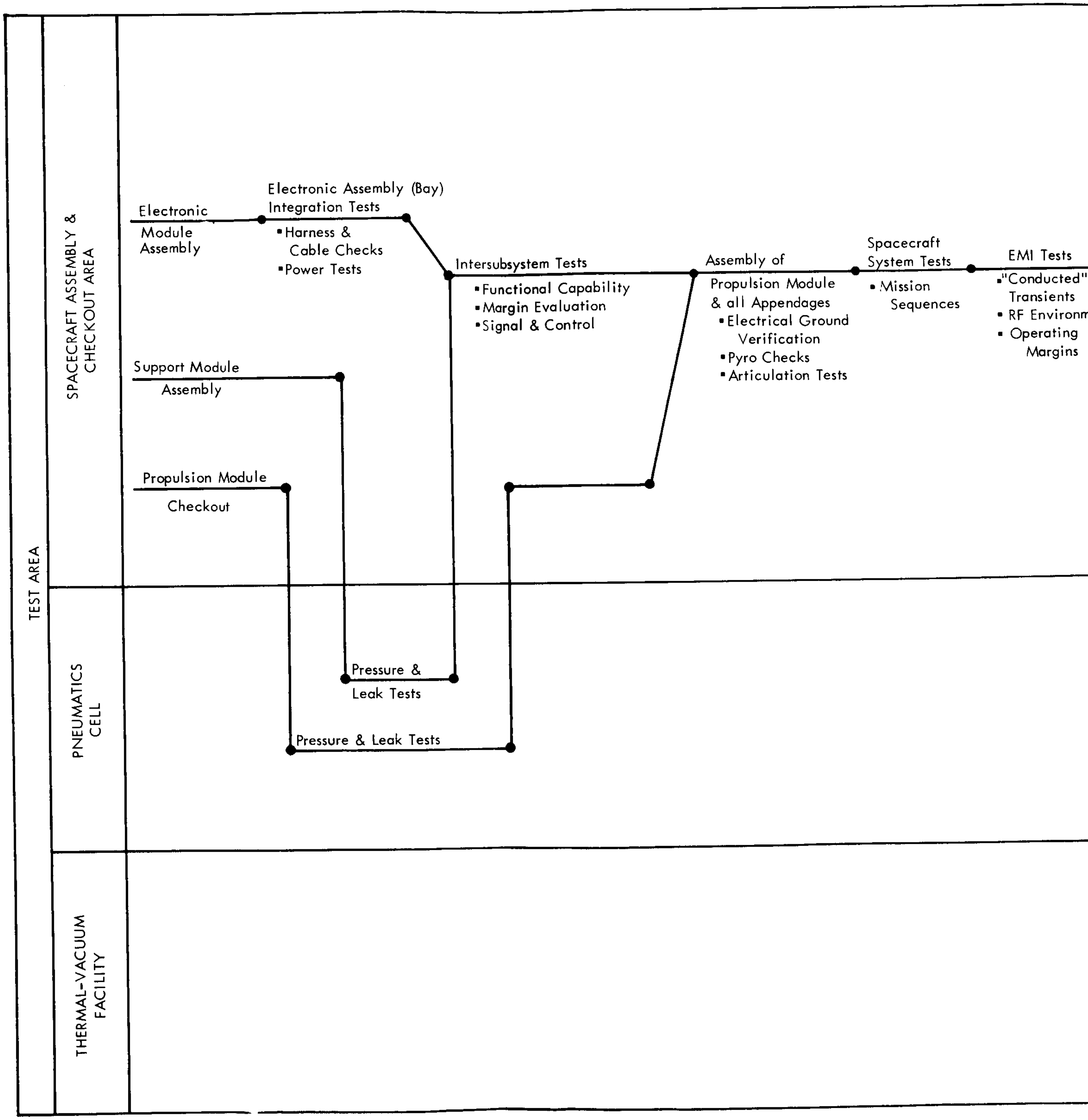

3-3A
Foldout Frame

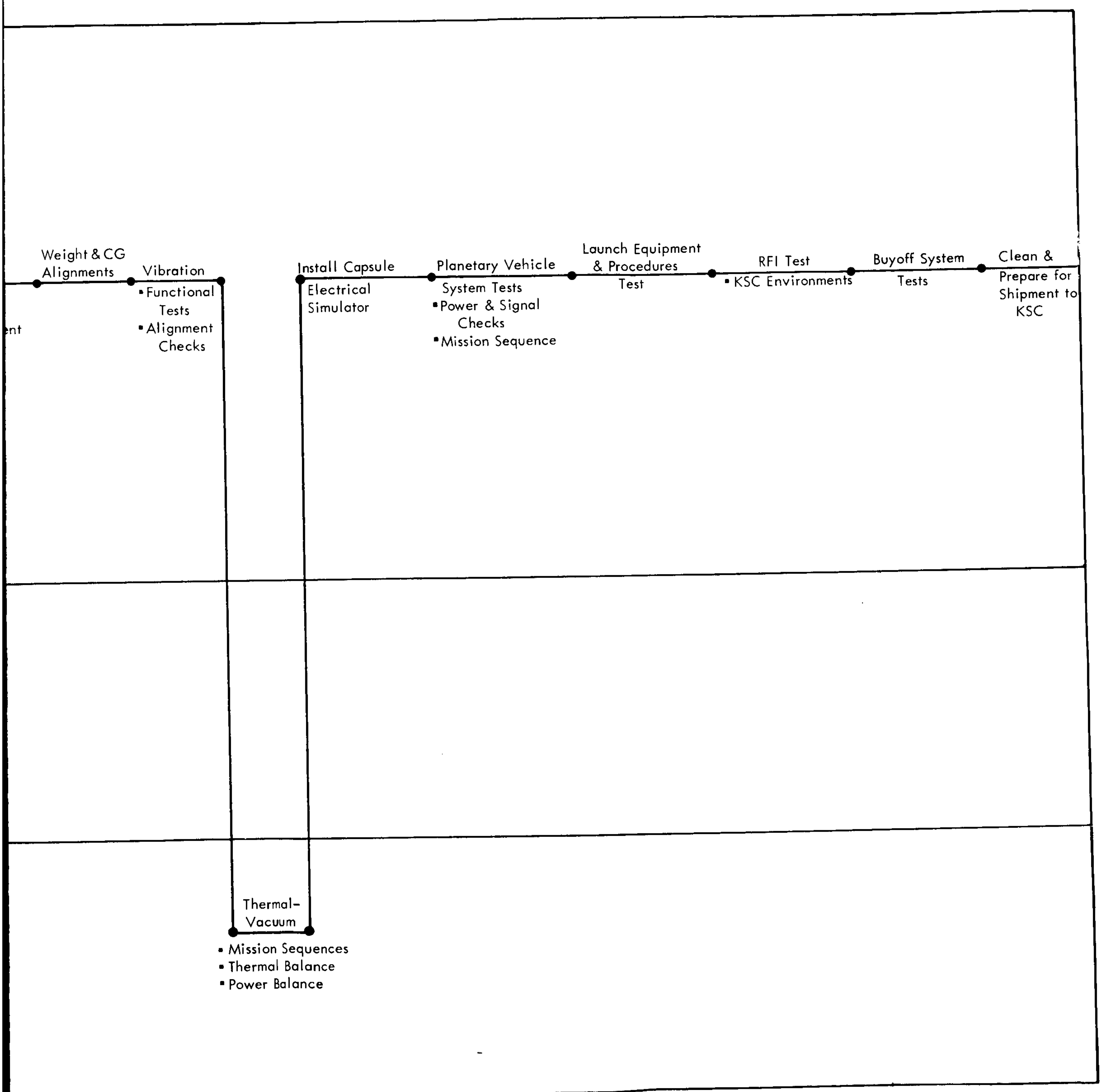

Figure 3-2. Flight Spacecraft System Assembly and Checkout

Foldout Frame

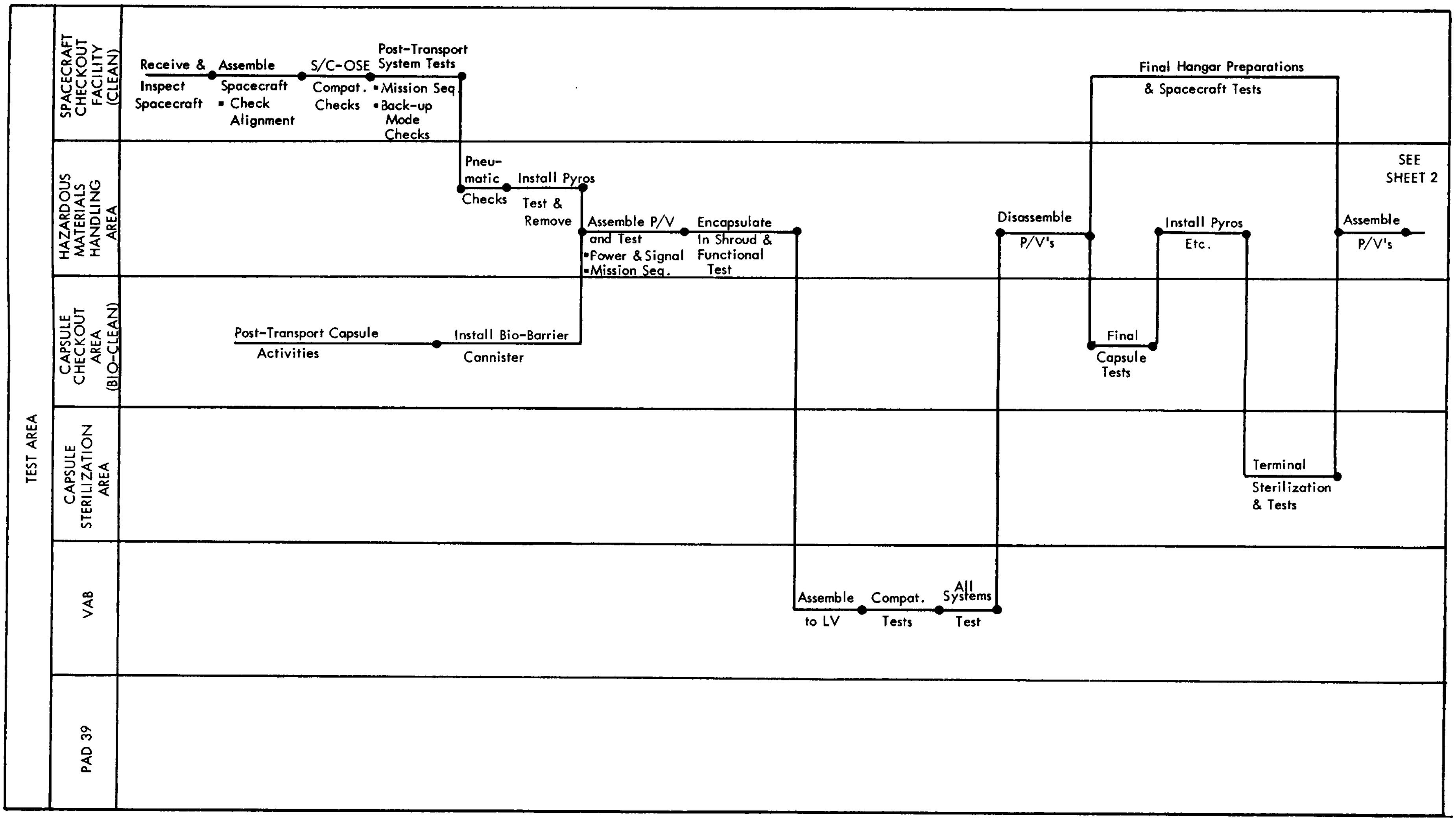

Figure 3-3. KSC Flow (Sheet 1 of 2)

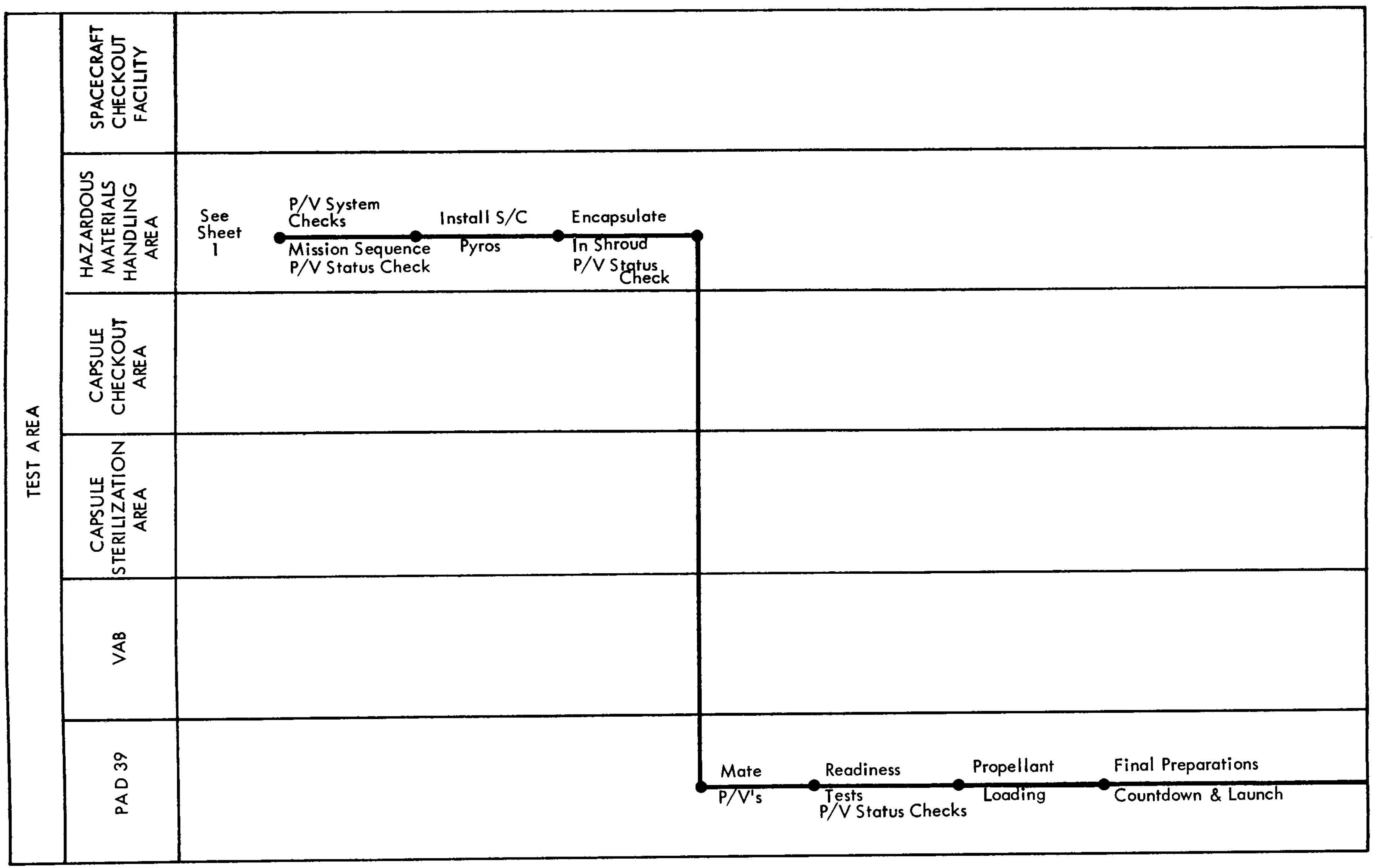

Figure 3-3. KSC Flow (Sheet 2 of 2)

SECTION 4

APPLICABILITY OF APOLLO ACE

4.1 INTRODUCTION: APOLLO ACE SUMMARY

The acceptance checkout equipment - spacecraft (ACE-S/C) is the integrated checkout and display system (providing for centralized programmed control of spacecraft checkout operations) developed for the Apollo spacecraft. The ACE-S/C ground station can accommodate independent subsystem and/or integrated system testing in either manual, semiautomatic, or automatic operational modes. Large quantities of test data can be processed and displayed in real time as well as recorded for post-test analysis.

ACE-S/C ground stations are installed at the following locations for the purposes indicated:

a. Merritt Island Launch Area (MILA), Merritt Island, Florida. ACE-S/C is utilized to conduct the complete range of subsystem and integrated system testing from receipt of the spacecraft modules through spacecraft/launch vehicle mating and final prelaunch checkout.

b. Manned Spacecraft Center (MSC), Houston, Texas. ACE-S/C is utilized in the testing of spacecraft and astronauts in simulated space and lunar environments. Reliability assessment data will be derived from extended spacecraft operations at MSC.

c. Grumman Aircraft Engineering Corporation (GAEC) Facility, Bethpage, New York. ACE-S/C is utilized for subsystem and integrated system testing of the lunar excursion module (LEM) during the various stages of factory assembly and checkout.

d. North American Aviation Corporation (NAA) Facility, Downey, California. ACE-S/C is utilized for subsystem and integrated system testing of the command module (CM) and service module (SM) during the various stages of factory assembly and checkout.

The ground station part of ACE-S/C operates in conjunction with the spacecraft during checkout via the digital test command system (DTCS) and the digital test measurement system (DTMS). The DTCS receives, decodes, and converts digital test commands from the ACE-S/C ground station; and the DTMS codes and multiplexes the response of the spacecraft subsystems to digital commands for transmission to the ACE-S/C ground station for processing.

The ACE-S/C ground station is divided into three primary areas: the computer room, the control room, and the terminal facility room. The computer room contains the data acquisition subsystem, the computer subsystem, and data transmission equipment. The control room (Figure 4-1) contains consoles, data display system, and part of the command generation equipment. The terminal facility room contains patching facilities, parts of the command and data display subsystems, and timing equipment.

Specifically the ACE-S/C ground station can perform the following functions:

a. Provide the control, display, data processing, and recording required to control spacecraft stimuli equipment.

b. Receive, process, display, and record spacecraft parameter data derived from spacecraft ground and flight telemetry systems.

c. Provide self-check and calibration capacity for itself and related equipment.

The ACE-S/C system was developed to checkout a broad range of complex spacecraft. In its present Apollo application, ACE-S/C is being utilized to perform the complete range of spacecraft testing from factory checkout to launch. In accomplishing this, the ACE-S/C system:

a. Completely tests integrated systems and independent subsystems.

b. Provides any desired degree of test automation.

Figure 4-1. ACE-S/C Control Room

c. Processes large quantities of data for real-time display.

d. Adapts to any system or test mode by means of software changes.

The following additional advantages are accrued through the utilization of ACE-S/C on Apollo:

a. Provides for astronaut participation and launch control.

b. Tests across system interfaces.

c. Uses flight telemetry equipment.

d. Minimizes cabling by serial transmission of commands and data.

e. Has built-in self-check capability.

Appendix A, Apollo ACE Summary Description, provides a block diagram and a brief description of the function of each element for a typical ACE ground station. Facilities required to support ACE are also included.

4.2 VOYAGER ACE APPLICATION

The Voyager Program would use the ACE equipment generally in the same manner as Apollo. There are, however, several differences which have been identified, and are highlighted in the following sections. The most significant of these are summarized below:

a. The spacecraft telemetry may have to be decommutated in the downlink computer rather than in the Radiation decommutators.

b. Ten-bit uplink and downlink data is desired to support trend tests rather than Apollo's eight-bit data.

c. A large amount of high frequency data analysis (wave shape analysis, noise measurements, solenoid valve signatures, etc.) will be done at the spacecraft interface equipment (SIE).

d. R- or C-STARTs (selections to actuate random testing) will be used to generate spacecraft commands rather than just to control baseplate units.

e. Limits used within the decommutater will be kept current, and for every uplink command at least one set (if not several) of downlink limits will be changed; this is not currently done on Apollo but the capability exists.

f. Only the current running automatic program and its alternates will be in computer core storage at any instant. (If the Apollo philosophy of having all automatic routines in storage at all times were used, the ACE memory would be overloaded.)

g. Two locations (Radio Subsystem SIE and mobile launcher SIE) will be controlled and monitored during launch operations rather than just the mobile launch facility.

h. Parallel hardwire controls and monitors for critical functions from the launch control center (LCC) will be required.

i. Tie-in with Deep Space Station (DSS) 71 will be required.

j. Differences between system test and launch requirements are a large step function, whereas with Apollo they are quite similar.

Figure 4-2 illustrates the use of Apollo ACE equipment for system tests and Figure 4-3 for pad operations. The major elements are discussed below:

4.2.1 SPACECRAFT INTERFACE EQUIPMENT

The spacecraft interface equipment (SIE) for system tests is summarized for each subsystem in Appendix B. This equipment is peculiar to the Voyager spacecraft and is operationally located within 100 feet of the spacecraft during systems test. Its basic functions will be required regardless of what approach is used for operational support equipment (OSE). (Essentially no Apollo spacecraft interface equipment has been identified that can fulfill the Voyager requirements. This was expected since the original ACE concept was that this equipment would have to be tailored to each new type of spacecraft.)

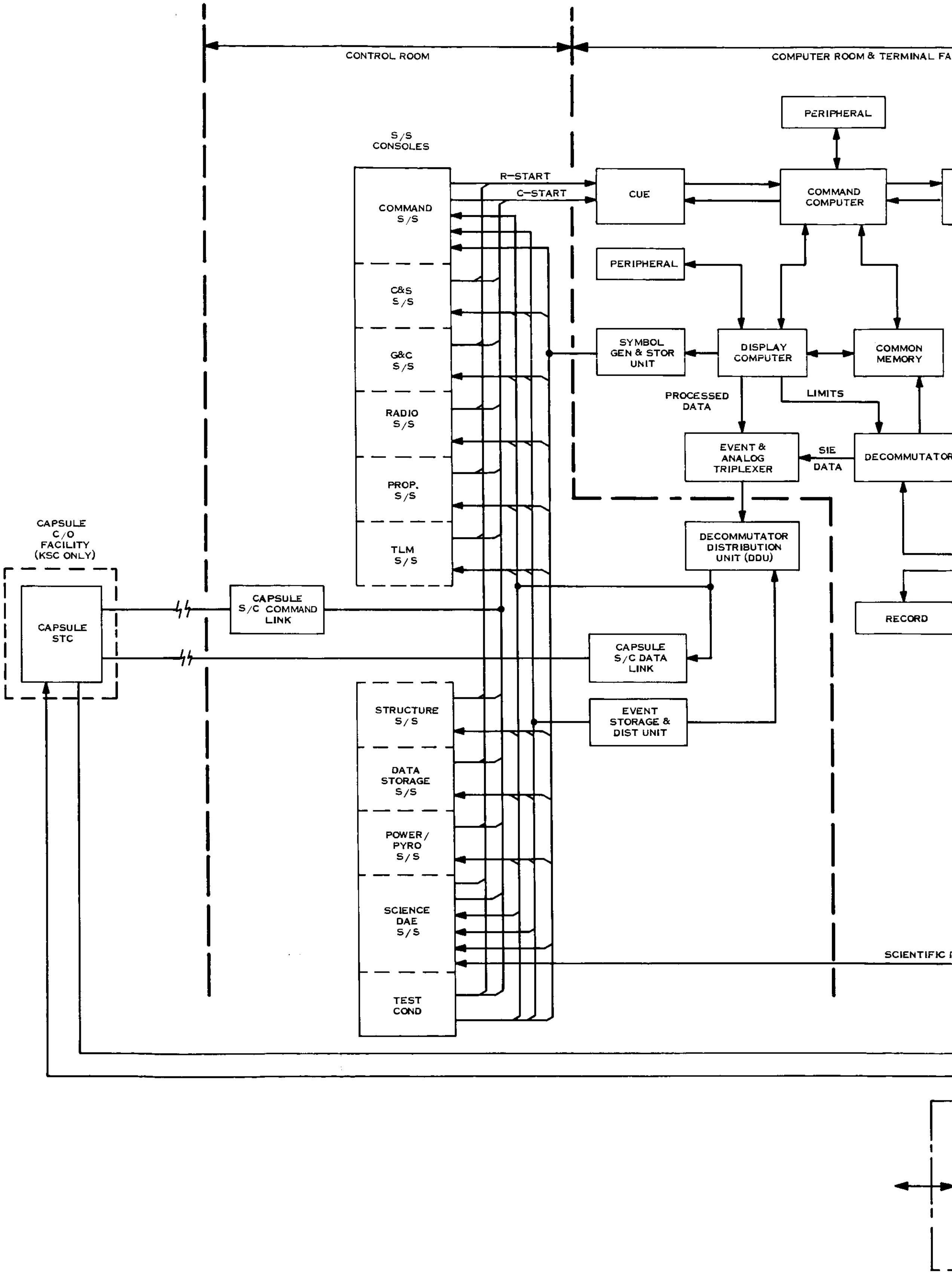

Figure 4-2. Voyager ACE System Test Configuration, Simplified Block Diagram

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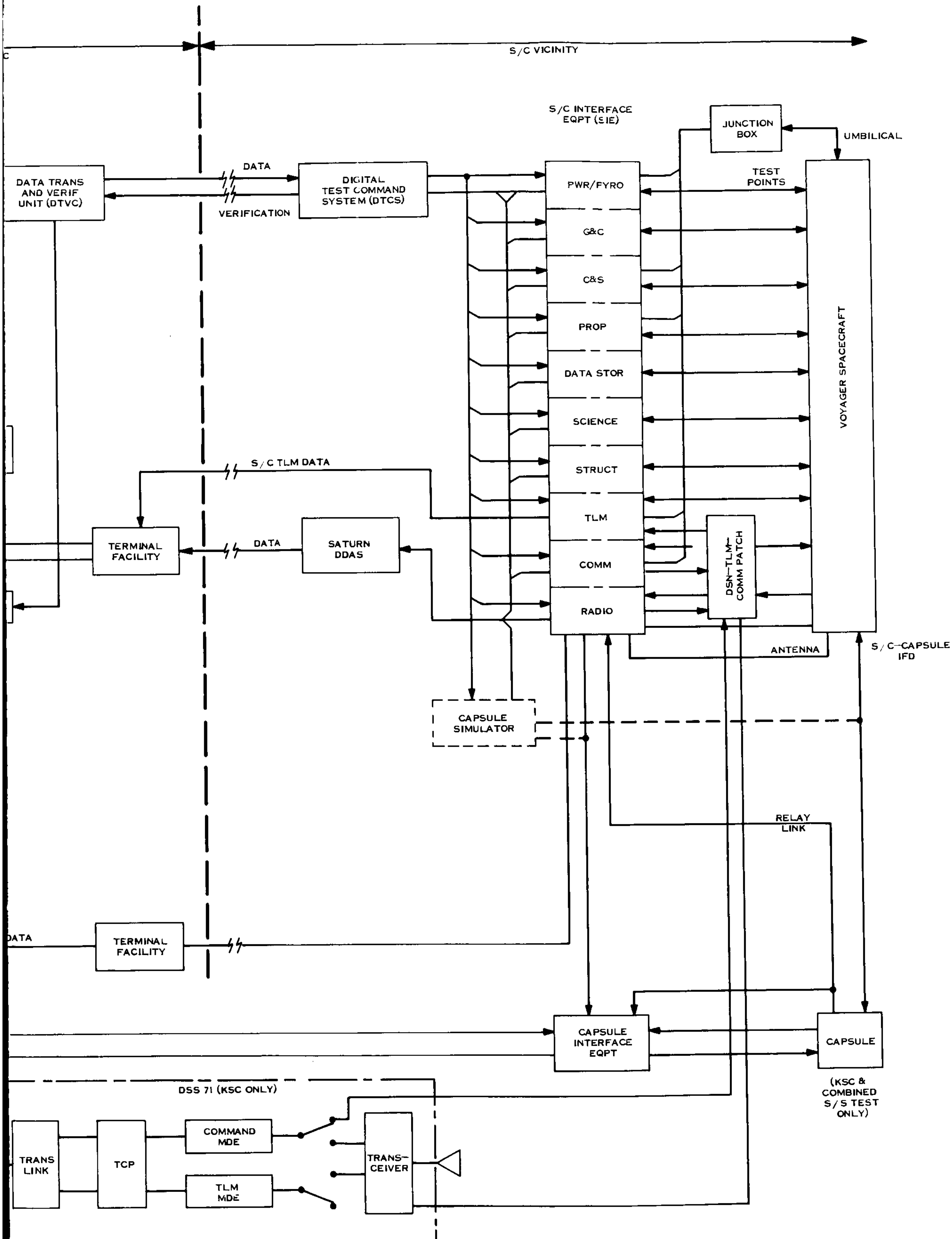

Figure 4-2

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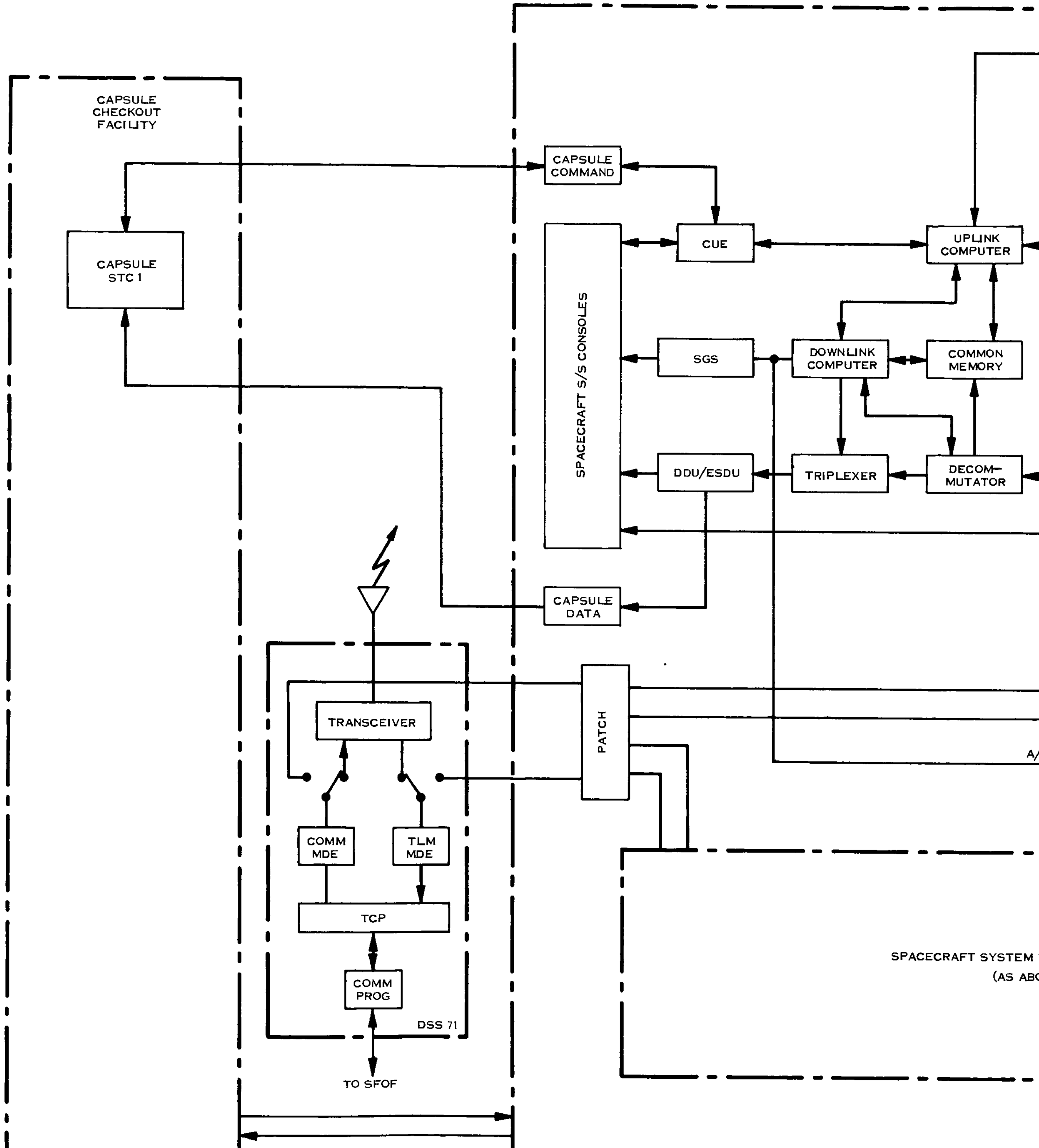

Figure 4-3

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4-6 B

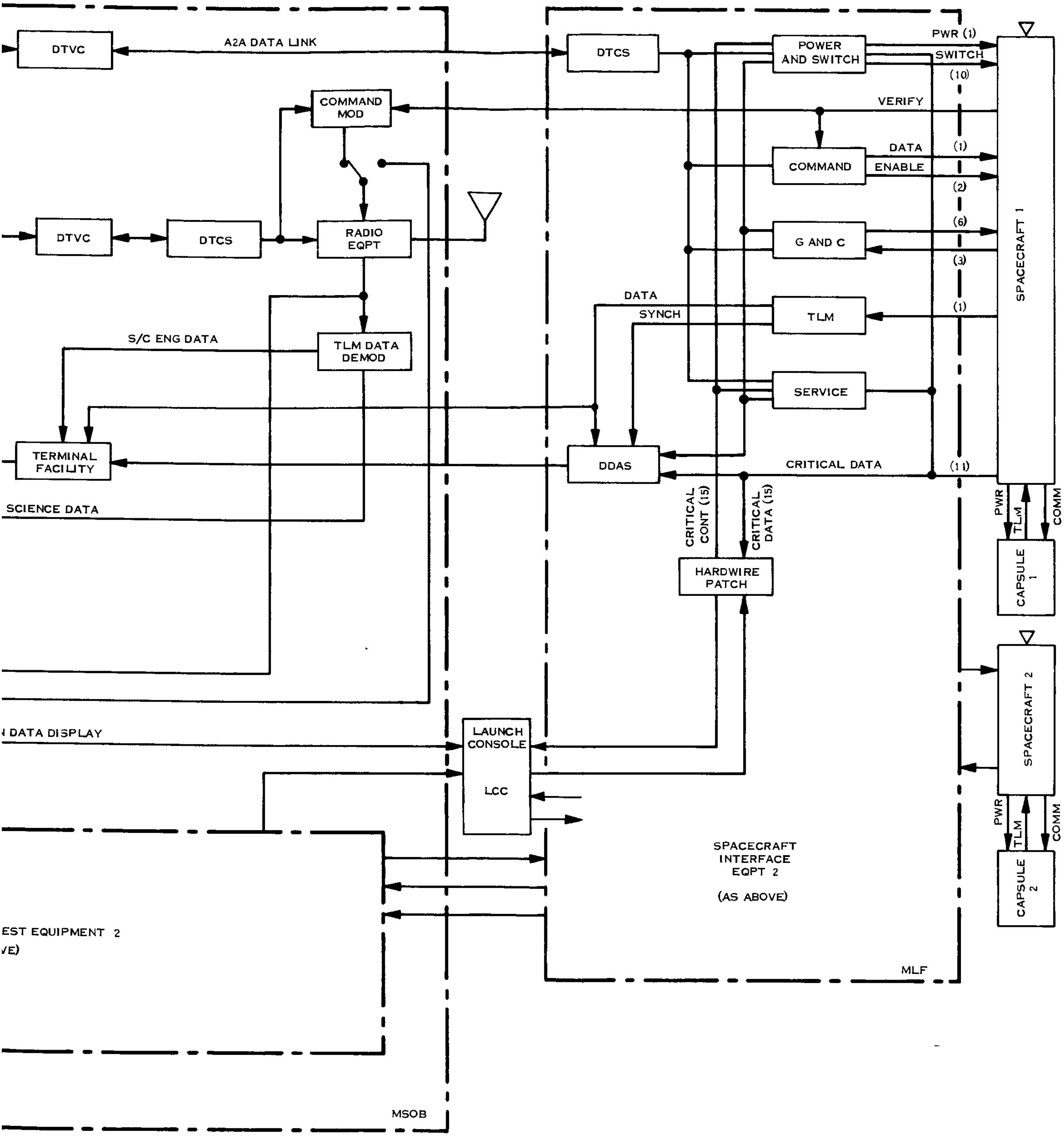


Figure 4-3. Voyager ACE Launch Mode, Block Diagram

The Voyager trend test concept requires that in order to detect spacecraft parameter trends, the measurement and stimulation uncertainty contributed by the OSE must be minimized to the degree that it be small compared to the expected spacecraft trends. While the Voyager spacecraft variations are not known today, historically trend test controls and measurement accuracies should approach the state of the art. Apollo ACE is an 8-bit system (1 part in 256 or 0.4 percent); at least a 10-bit system (1 part in 1024 or 0.1 percent) is desired for Voyager.

Interfacing the Voyager SIE equipment with ACE--using the maximum amount of ACE equipment and software--requires this equipment to be controlled by ACE's DTCS baseplate units (outputs contact closures or digital-to-analog (D/A) conversion). There should be no major problem in the majority of cases in obtaining the greater accuracy (or repeatability) required in the stimulation of the spacecraft in order to meet the trend test concept. This is true since the intent is to repeat the stimulation from test to test--this can be done in many cases by using, in effect, the eight most significant bits of a 10-bit accurate stimuli generator as controlled by eight contact closures, or by using an improved D/A converter. This will cause the stimulation to be accurate and repeatable to 10 bits (0.1%) but with a restriction that the resolution be 1 part in 256 rather than 1 part in 1024. Where this is not satisfactory (an example is finding the trip point of a threshold detector by gradually increasing the input until the output changes state), one of two techniques can be used; i.e., (1) one 8-bit word can be used to set the eight most significant bits of 10-bit (or more) up/down counters which can be incremented or decremented by another control for the least significant bits, or (2) the ACE guidance and navigation (G&N) buffer can be used to shift out into an SIE register its 16-bits which can in turn distribute 10-bits throughout the SIE as controlled by 6-bit address scheme included in those data bits. This last mode has the disadvantage of not making full use of ACE's normal verification path from the DTCS. This will have to be compensated for by incorporating similar verification circuits between the SIE buffer and its destinations and using the downlink OSE data as a final check.

The Voyager trend test requirements introduce two major problems into the downlink data obtained from the SIE or from the spacecraft direct access (hardwire) leads. The first of these is the desire to do wave shape analysis, and observe noise on practically all of the direct access leads. This type of analyses is not done continuously on most of the leads--the notable exception being the RF signals--but only periodically. This type of analysis is incompatible with any remote control and display scheme that commutates the data. One way around this problem is to be able to stop the commutator on any one selected point for a desired period and return only that one signal to the remote location for that period. Another way is to have the commutator scan its normal inputs and insert in between each point a sample of the high frequency data. The ACE's DTMS maximum sampling rate (i.e., stopping the commutator) is 5000 samples/seconds. This is not high enough to detect the parameters of interest. The remaining alternate is that this periodic high frequency data analysis will have to be performed locally at the SIE.

The second problem is that ACE's DTMS is an 8-bit system while a 10-bit system is desired. Since the Apollo ACE signal conditioners, multiplexers, and analog-to-digital (A/D) converters are designed to meet an 8-bit system accuracy, the choices available are to upgrade these components to a 10-bit accuracy, design a new system, or use an existing 10-bit system.

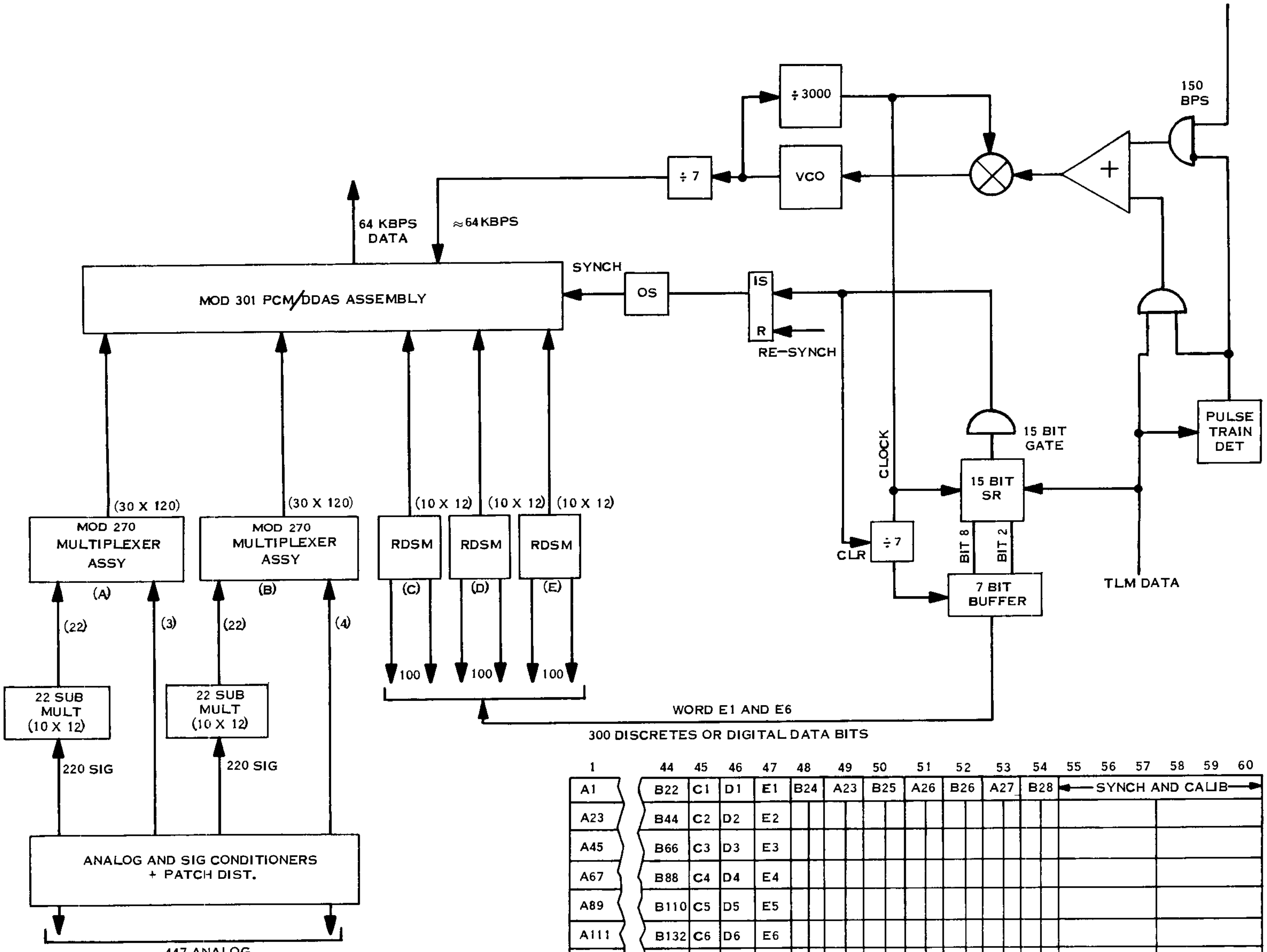
The first two choices are essentially the same since to upgrade the Apollo ACE components, so many elements must be changed that, in reality, a new system has been generated. This approach cannot be justified if there exists an "off the shelf" system that it is essentially compatible with the Apollo ACE requirements. The main effort was, therefore, to identify such a system. This search ended with the Saturn ESE digital data acquisition system (DDAS).

This DDAS is a 10-bit system, used on the Saturn program, and appears to be essentially compatible with the Apollo ACE. The exception is that the digitized data is placed on a 600-kc carrier for transmission via coaxial line to the remote control and display area where the digital data must be recovered from the carrier. ACE takes the digital data back directly on an A2A transmission system. Investigation indicates that the digital data is available as an output from the DDAS directly and this point can therefore be tied into the links as now done by the Apollo ACE equipment. Figure 4-4 shows a possible Voyager ACE DDAS system and format. As discussed in more detail in the Decommutation section (Section 4.2.2.2), there are significant gains in maximizing the use of existing ACE hardware and software if the spacecraft's telemetry data is interleaved into the OSE and direct access downlink data. Interleaving requires that the two data streams be synchronized as to bit rate, and the source rate must be the spacecraft's telemetry rate. The Task D telemetry form will be 7-bit words transmitted at 150-bits/seconds in one of several commanded formats. The bit synchronizing of the two streams can therefore be implemented by driving the DDAS system at a rate such that the telemetry word appears once every 300 DDAS words, or at a rate equal to (3000 x 150/7 $\approx$ 64,000 bits/second. The normal DDAS data rate required to handle the Voyager requirements (approximately 400 analogs and 300 discretes) would be 72,000 bits/second derived internally by the Mod 310 PCM/DDAS assembly (note: Saturn requirements make the rate 144,000 bits/second). While this external drive capability does not exist today in the Mod 310, it has been estimated as a simple task to accomplish. If the ground station can decommutate the resulting interleaved data in the Apollo ACE manner, the savings in software would more than offset this cost. It must be noted that this interleaving approach is required only if the docummutator can handle the final telemetry format interleaved into the ground data. If the spacecraft telemetry is as in Task B, and the decommutator can not handle the format, then interleaving to this extent would not offer any significant advantages. In that case the telemetry data can be sampled as a digital word at a rate higher than it occurs.

4.2.2 COMPUTER COMPLEX

4.2.2.1 Computers

The Apollo ACE computer configuration of two CDC 160G computers is an example of how current state-of-the-art improvements would preclude that configuration if ACE were to be



1	44	45	46	47	48	49	50	51	52	53	54	55	56	57	58	59	60
A1	B22	C1	D1	E1	B24	A23	B25	A26	B26	A27	B28	← SYNCH AND CALIB →					
A23	B44	C2	D2	E2													
A45	B66	C3	D3	E3													
A67	B88	C4	D4	E4													
A89	B110	C5	D5	E5													
A111	B132	C6	D6	E6													
A123	B154	C7	D7	E7													
A155	B176	C8	D8	E8													
A177	B198	C9	D9	E9													
A199	B220	C10	D10	E10													

Figure 4-4. DDAS Equipment and Possible Format

designed today. A tradeoff study was conducted to establish whether Voyager required a different computer or, since approximately six sets are required to support factory tests, possible cost savings could be achieved by using a modern computer.

Computer loading studies to date indicate that Task B Voyager's worst case will not exceed the present computer configuration. Consequently, since Voyager's operational support equipment reliability, availability, etc., are not more stringent than Apollo's, the choice comes down to potential saving versus schedule risks of incorporating a new computer complex into the ACE configuration.

Using a more modern computer and adapting it to the ACE input/output (I/O) paths would offer per-unit hardware cost savings. This saving is offset against the cost of developing a new design which entails hardware design, and development, software design, verification testing, and installation and removal costs at KSC. In order to assess the magnitude of the net cost differential which could be realized from a more appropriate computer selection, rough-order-of-magnitude (ROM) costs were generated. These costs indicated that the maximum saving to the Voyager Program might be as high as $6 million if six new computer complexes were required and maximum advantage was taken of the existing ACE software techniques, etc. This "saving" would disappear, however, if NASA made available two existing ACE stations to the Voyager program or if entirely new software or I/O techniques had to be generated. It is believed that even this potential saving is not significant when compared to the risks involved to the Program by replacing the heart of an existing and proven hardware and software system with one that does not exist today.

The question of whether only one CDC 160G per ACE computer complex should be used rather than two was considered. This was resolved when it became apparent that the expected loading would be higher than desirable (> 75 percent at this stage of implementation) and that a great deal of the existing software would have to be redone. The maximum net gain would not be worth the risk, especially when one of the risks is an overloaded computer.

The recommendation based on these studies is that the Apollo ACE CDC 160G computer configuration should be maintained on any additional ACE system which might be procured for the Voyager Program.

4.2.2.2 Decommutation

The configuration using Radiation 540 decommutators and its input/output circuitry remains the same at KSC. A possible physical difference for factory use would involve the installation of only one decommutator (as it is considered a standby and not required by the system to be "on line").

The only hardware change required is that required to accommodate the 10-bit OSE or DDAS telemetry data words. The fact that the Apollo ACE stations can handle completely only 8-bit data requires a change. The proposed scheme for accomplishing this is for the decommutator to perform the following functions with the limitations as noted:

a. Establish bit, word, frame synchronization.

b. Tag data words with an identification code stored in the decommutator by the downlink computer.

c. Compare the eight most significant bits to high and low limits as stored in the decommutator by the downlink computer. Tag the words if they are out of limits.

d. Compare the current eight most significant bits to the corresponding data point received earlier. If the change is greater than an amount stored in the decommutator by the downlink computer, tag the word.

e. Output the eight most significant bits and identification (ID) code (b) via the event and analog data triplexer. Note: This means the analog data to be displayed on meters or strip chart recorders will have 8-bit accuracy (approximately 0.5%) which is more than enough for that type of display. This also means that words containing discretes or digital data that are to be displayed via this path must use only the eight most significant bits of the 10-bit word.

f. Transfer the 10 data bits plus the tag from (c) and (d) to the computer memory where it will be processed/recorded for later trend analysis.

The existing decommutators perform all of the above with the exception that it transfers only eight data bits in (f) above. Since there are two spare bit locations in the interchange word used between the computer and decommutator for that function, it is not a large problem to switch the two added bits to those locations.

It should be noted that when synchronized, interleaved data is present, both the OSE data and the spacecraft data are being decommutated by the same unit essentially simultaneously.

During the early phases of the study when the Task B spacecraft telemetry scheme was being used as a baseline, there was an incompatibility with the Radiation decommutators in that they could not handle the nine nonsynchronized medium and low speed telemetry decks. This is the only case discovered to date of a major part of ACE, expected to be "universal," that could not perform the Voyager task. (Note: the Task D spacecraft telemetry approach,while it uses synchronized decks when interleaved, may still be incompatible with the Radiation decommutators due to the widely varying data rates.) Because of this, a different telemetry decommutation approach would have to be provided for Voyager's Task B telemetry implementation. The alternatives considered were:

a. Modify existing Radiation decommutators.

b. Use ACE's downlink computer to decommutate the telemetry data as well as performing its other functions.

c. Provide a small general purpose computer for the decommutation function in place of the Radiation unit.

With the second of the above three alternatives using the downlink CDC 160G for decommutating, the main question is, "What is the added loading effect of this telemetry data decommutation on the downlink computer?" If the computer can handle this new load as well as maintain its other functions with an adequate margin of safety, then the existing downlink computer alternative should be used. Likewise, if an adequate margin of capacity is not available, then the choice must be one of the other alternatives. Since it is unrealistic to make the major degree of modifications required to give the Radiation decommutators the capability required, the choice would be narrowed to that of adding the small general purpose computer.

The loading effect of the decommutation on the downlink computer was estimated (as well as an estimation of the load due to its other operations using extrapolated Apollo loading figures). The results were as follows:

a. Performance of decommutating functions as above for the Task B telemetry format with a data rate of 17 words/second will occupy 0.5 percent of the computer's time.

b. Assuming that the Voyager processing requirements (converting to engineering units, formatting and outputting to alphanumeric (A/N) displays, recording compressed data, etc.) are comparable to Apollo's per-word rate, and assuming Voyager's data rate (ground data and telemetry data) is 640 words per second, this will require 10 percent of the computer's time.

c. Inputting and outputting load and handling the ADAP routines for Voyager as in Apollo will require 35 percent of the computer's time. The total load, therefore, is 45 percent of the computer's time.

This load is satisfactory at this time and the use of the downlink computer for decommutating the telemetry data is an adequate solution. The net effect of this is that there is no hardware change required, but a considerable amount of new software will have to be generated to perform the decommutation of the spacecraft telemetry data. It should be noted, however, that as the telemetry data rate is increased, the load increases essentially linearly and can therefore approach overloading the computer if its data rate is increased to the region of 1500 words/second. The effect of data compression and error coding cannot be established since this will vary greatly depending on the technique used, but either one may cause excessive loads. Each of these, therefore, may lead to the choice of a general purpose computer for decommutation or perhaps even some special purpose equipment (primarily for error coding) may eventually be required for Voyager.

The study was also extended to the decommutation of the OSE data stream (SIE data and hardwire data) by investigating whether this data can or should also be decommutated by the downlink computer. If this were done, the Radiation decommutators could be eliminated. The added load on the downlink computer to do this decommutation would require 25-45 percent (depending on rate of change of the data) of the computer's time. With this added on to the 45 percent obtained from the telemetry load, figures would bring the total computer load to 70-90 percent. At this stage of developement and with the lack of detail knowledge, this loading is much too high (50 percent is considered a maximum safe figure). The use of the downlink computer for decommutating all of the data is therefore not an acceptable alternative.

The Radiation decommutator's capability has been bypassed by state-of-the-art improvements since its design. Its relatively high cost also makes it questionable as to whether the small general purpose computer approach might not be the more attractive alternative.

ROM costs were generated which indicated that a maximum potential program cost savings of $400,000 might be achieved by using a small general purpose computer for all of the decommutation functions in place of the Radiation decommutators. This figure can be reduced to zero, however, if NASA can make two Radiation decommutators (modified) available to the Voyager Program. Even the maximum difference is not felt to be too significant compared to the intangible advantages inherent in maintaining a working system and increasing the assurance of meeting the launch window. Therefore, if the Task B requirements are maintained, the preferred solution to the decommutation problem is to use the Radiation decommutators for the ground data and the downlink computer for the telemetry data obtained as a ground data word from the Radiation decommutator.

It should be noted that if the effect of future studies is to radically change the telemetry format or data rate such that the downlink computer becomes overloaded, the small general purpose computer may be required. If it is required, then it should also be used for the ground data and the Radiation decommutator eliminated. It should further be noted that the

Task D scheme may have just the opposite effect and may allow the entire task to be done with the Radiation decommutators exactly as in the Apollo application.

4.2.3 CONTROL AND DISPLAYS

4.2.3.1 Apollo ACE Configuration for Voyager

Using the control and display requirements extracted from Appendix B, a grouping of existing ACE consoles was achieved such that each subsystem's requirements was satisfied by adjacent highboy consoles and, if required, a lowboy immediately in front of the highboys. This group required 22 highboys (25 in an Apollo group) and 8 lowboys (23 in an Apollo group). Because the following ground rules were used, this is considered a worst-case analysis with the existing requirements:

a. The capability for each subsystem engineer to cause the spacecraft Command Subsystem or Computer and Sequencer (C&S) Subsystem to execute a spacecraft command that controls his particular subsystem was to be provided by means of R-STARTS on a one-to-one basis for each command. (Task B spacecraft contains 213 command functions, most performed by both the Command and the C&S Subsystems.)

   The data flow for this usage of R-STARTS is: (1) R-START to uplink computer which looks up the proper command format for the particular R-START operated. (2) This format is outputted via the DTVC to the command modulator in the SIE, which combines the data with the pseudo noise (PN) synchronization code and sends this to either the Radio Subsystem SIE or directly by hardwire to the spacecraft Command Subsystem. (3) The spacecraft Command Subsystem operates on the PN code and data and outputs the command, causing the control requested to be executed.

b. All of the data was to be displayed on meters and lights. Where the existing consoles did not have the capability to time share meters, as is often desirable, additional meters would be required.

Both of the preceding create maximum requirements that can be radically reduced, if necessary. The use of R-STARTS to call up commands is extremely valuable from the standpoint of operator's convenience, speed of operation, trouble-free operation, etc., but one C-START can do the entire spacecraft command call-up task, if required. Its drawback is that it is not convenient to use since the operator must always look up the proper C-START code before he can be assured of executing the proper command. However, by judiciously selecting which commands should be immediately available and therefore initiated by R-STARTS and which could be made less convenient to operate and be initiated by C-STARTS, a large reduction (about 100) in the number of R-STARTS could be achieved.

The display requirements can also be drastically reduced by time-sharing the meters for similar readings that are not required simultaneously. This time sharing can be achieved as done currently by ACE, or by using the ID code outputted with the data by the decommutator as a means of identifying the meter it is to be displayed on rather than to identify the data. This scheme is suitable when the data is to be displayed at only one station or in identical manner if at more than one station. This should be the usual scheme. This can be accomplished as a response to an R- or C-START by having the uplink computer "command" the downlink computer to cause the decommutator to alter the ID codes as desired. The disadvantage of this scheme is that it requires more bookkeeping by the downlink computer. The display requirements can be further reduced by placing more reliance on the alpha-numeric display.

The net result of the preceding is that it is expected that the existing ACE consoles can quite comfortably satisfy the normal Voyager requirements.

It should be recognized that Voyager may have some peculiar requirements that will require local control and monitoring (at SIE) or require extensive processing equipment in the control room (see Science OSE, Section 4.2.4.1). There are peculiar requirements that can be implemented via the ACE consoles. Typical of the latter is the requirement for the Computer and Sequencer Subsystem engineer to generate PN codes 25 bits long (34×10^6 combinations) that correspond to time in seconds. Even though the majority of tests are preplanned such that he will know the desired PN code, he should have the capability to generate PN codes corresponding to any desired time. It is impractical for him to have a look-up table or for the uplink computer to perform the transformation (i. e., the maximum number would require the full time of the computer for approximately 10 minutes). Therefore, as indicated in Appendix B, hardware is required for performing this decimal to PN (and reverse) code transformation.

This is initiated by having the operator use a C-START to generate the desired decimal value: PN code length, etc. This data is sent to the SIE where the conversion is made to the PN code. The output is returned to the remote area by the OSE PCM telemetry, where the downlink computer outputs the data via the A/N display. The operator takes this data and inserts it in the C-START identified as the PN code to be sent to the specified Computer and Sequencer register via the spacecraft Command Subsystem. While this may not be the optimum method of eventually performing this task, it represents a solution to an abnormal problem making maximum use of Apollo ACE capabilities, and illustrates that there are methods for getting around apparent limitations.

4.2.3.2 Optimized Panel Configuration

The Apollo ACE panels were laid out to a large degree so that they could be applied on a universal basis. This was accomplished by designing a module of switches, a module of lights and a module of meters for each console, the number of modules provided, however, being that required to fulfill the requirements of that console.

There is nothing that shows the relationship and interaction of controls and displays. While the lights can be labeled to identify the condition being displayed, and the meters labeled and scaled in engineering units, with go/no-go limits, it is not immediately obvious to the operator what switch or control action is required to correct no-go situations nor can he readily verify that the displays are correct for the current switch settings. The result is that the human factoring aspect is low and the ACE operators must follow a "cookbook-type" operation, with each action and its response spelled out in detail. If something unexpected happens, there is nothing inherent in the OSE design to help the operator make the correct decision.

This is further compounded by the C-START control action which requires the setting of ten 12-position switches to the exact settings required to perform one operation. If these ten switches (12^{10} combinations) must be set quickly based on the operator's memory, faulty operations may occur.

This type of operator-panel interface would not be consistent with the proposed Voyager approach of having engineers who are thoroughly familiar with the spacecraft and OSE operate the controls so that they can perform immediate corrective action, carry out investigative type sequences, and perform troubleshooting operations. (Opposed to this is the obvious fact that these panels have been used on the Apollo Program and they can, therefore, be made to work on the Voyager Program.) The question then becomes how much is human factoring worth and how much would it cost to incorporate human factoring such that the operator is made aware of the interaction between controls and between displays and controls.

This would be accomplished by locating controls and displays with respect to each other, displaying data flow paths, identifying legends, and establishing mnemonic coding of controls, etc. The degree of human factoring desired would be, as a maximum, that used in "process control" type displays in which essentially an instrumented schematic or block diagram is presented. The minimum would be panels typical of the Saturn ESE.

What is required for any human factoring (and is not currently possible with ACE) is the capability of mounting switches, lights, and meters any place on the operator's panel. In order to achieve this while still providing flexibility at a reasonable cost, the panels alone should be all that have to be changed in order to update the controls and displays. This requires that all the electronics be a part of the console proper and laid out and cabled such that they can accommodate various numbers of displays and controls. The panels would plug into the cabinet cabling to pick up the desired meters and light drive circuits and switch logic functions. Therefore, changing the layout would require only the cost of generating a new panel, i.e., plate, lights, meters, switches, silk screening, and back cabling, which would be plugged into the console cabling to the existing logic. Using the "process control" approach, this change is required only when the spacecraft block diagrams or their interfacing OSE are changed, which should be infrequent on a program such as Voyager.

The major obstacle to accomplishing this random placement of panel components using existing Apollo ACE equipment, are the START modules. They are packaged as a group of switches integral with their associated electronics. As such, the switches cannot be placed at random on the panels and, therefore, prevent human factoring. The R- and C-STARTS also give only one type of control, that is, set pattern desired (R-START set four 2-position switches, C-start set ten 12-position switches), then execute pattern by operating an execute switch for that particular module. Other forms of switching are also desired, such as single switch operation (momentary on-off, push/push), thumbwheel switches, a digital potentiometer type control and a keyboard (to insert near-English mnemonic codes into the computers). However, by redesigning the switch circuitry such that it can be remote from the switch and by incorporating designs to accommodate the other types of switch operation--maintaining the existing interface between the START modules and the J-boxes and command unit executor (CUE)--minimum human factoring could be accomplished.

As preliminary layouts and cost estimates indicate that this should not be difficult or costly to accomplish to the extent outlined above, modifications to at least this extent are recommended to assure compatibility with the intended Voyager operational approach.

4.2.4 VOYAGER-PECULIAR INTERFACES

4.2.4.1 Science OSE (Including Data Automation)

The Science OSE is largely undefined at this stage and therefore represents an indeterminate factor in this application study. It is, however, believed that their normal controls and monitors can be made compatible with ACE and their equipment integrated into the SIE and into the control room. It is expected that they will have some additional science-peculiar requirements such as the display and analysis of spacecraft-generated television signals in the control room. This equipment will not be ACE type equipment but will probably be similar to that used in the Deep Space Network's Space Flight Operations Facility for producing and analyzing pictures returned from Mars.

Provisions have been made (as shown in the overall block diagram, Figure 4-2) to return on one separate line all of the Science Subsystem's scientific data. This data, in GE's Task B spacecraft, is essentially the Data Storage Subsystem's recorded data playback. This data will be sent directly (after demodulation by the Radio Subsystem SIE) to the Science equipment unprocessed. The Science equipment will have to recover their data from the error-correcting encoding scheme employed by the spacecraft's Data Automation Subsystem. This will probably require a relatively complex piece of special purpose equipment. This data must be correlated in real time with the real-time spacecraft telemetry data which will identify which tape unit is being played back at that instant. This data (spacecraft telemetry and test points, and OSE) will be given to the Science OSE via the normal routing of data (i. e., decommutator to computer and to the event and analog data triplexer).

As indicated in Appendix B, estimates of Science Subsystem SIE controls and monitors have been made. This, together with the spacecraft command and telemetry list, constitute the worst-case display and control requirement for the consoles, and was used to establish that the Apollo ACE's control room was compatible for Voyager. The Science Subsystem will also require floor space for its error-correction decoder and TV display. (The area presently occupied by most of the lowboys can probably accommodate this; or if this equipment becomes too large for that area, it can be incorporated into the SIE.)

4.2.4.2 Radio OSE

The Radio Subsystem OSE is different from the remainder of the OSE due to the relatively large amount of manual control and high frequency data that must be displayed for visual analysis. These cannot be accomplished conveniently remotely and will therefore be done locally at the SIE.

The basic problem is how to treat the spacecraft telemetry data pertaining to the Radio Subsystem and how should spacecraft commands--with the Radio Subsystem as their destination--be requested to be sent. These are both normally control-room-only functions. The question is: Should they be kept within the control room and have two Radio Subsystem engineers (one at the SIE operating only its controls and hardware monitors, the other in the control room operating on the spacecraft telemetry data and spacecraft commands affecting the Radio Subsystem, and perhaps some remote control functions and monitors of the SIE as well) or should the control-room-only functions be brought out to the Radio Subsystem SIE such that one operator can perform the entire task?

Each of these approaches has disadvantages, i.e., in one case, two operators at different locations perform interacting operations with incomplete knowledge; while in the other case, one operator is remote from all the other subsystem engineers and additional hardware is required. (Note: the additional hardware required is essentially the same as that described in Section 4.2.4.3 to support the capsule.) Since there is no ideal solution consistent with the ACE concept--unless of course the remote control room is in reality in the immediate proximity of the SIE--a preferred approach must be selected. The recommended approach, therefore, is to have two operators at two locations, with the control room operator in charge equipped with as much remote control and display capability as convenient, as is shown in the block diagram. (If the other approach were selected, there would be no problem in incorporating the required circuitry.)

A further complication of the Radio OSE is that it is to be used as a backup means during launch operations for obtaining spacecraft telemetry data and for inputting spacecraft commands, the prime mode being via data links and umbilical connections. Since the Radio SIE does not move to the pad with the spacecraft, additional links are required to tie it to the uplink and downlink computers. These links would tie into the OSE command modulator and the telemetry demodulator as shown in the launch block diagram (Figure 4-3), i.e., the uplink requires using an additional DTVC and the downlink can be accomplished by providing patch capability to allow patching the telemetry signal into the Radiation decommutator.

It should also be noted that these same points are capable of being patched into the Deep Space Instrumentation Facility's DSS 71 station.

4.2.4.3 Capsule OSE

The electrical interface between the spacecraft and the capsule consists of an inflight disconnect containing power, capsule telemetry data stream, spacecraft-originated telemetry synchronization data, spacecraft-originated command data (including data bits, alert pulses, end-of-word pulse, incorrect-command-sent pulse) and a relay RF link. While the flow of the capsule and its acceptance test, and its relationship to the spacecraft and its acceptance tests, have not been finalized, Field Flow (Figure 3-3) indicates that the mating of these two major elements would occur at KSC. There would follow compatibility and mission sequence type of tests, after which they would be demated and follow separate paths. They would then be remated and briefly reverified. The assembly would then be installed in the shroud along with another spacecraft capsule and the assembly would move to the pad for mating with the launch vehicle. (The pad operations, as far as this assembly is concerned, will have to be minimal.)

In order to support this flow, it has been assumed that the capsule must have its own system test equipment (STE) in its own checkout facility. The tie-in between the two STE's must be such that the following can be accomplished:

a. Capsule STE initiation of spacecraft commands that cause spacecraft power to be switched to the capsule.

b. Capsule STE initiation of either preprogrammed or operator-selected command data to be transferred from the spacecraft to the capsule.

c. Spacecraft OSE should recover the capsule data points from the spacecraft telemetry data stream and transfer that data unaltered to the capsule STE.

d. Spacecraft Relay Radio Subsystem OSE (SIE) should make available to the capsule SIE the detected capsule signals.

These can be accomplished by providing the following interface:

a. Uplink. The desired effect is for the capsule STE to have the ability to request the spacecraft STE to cause a spacecraft command to be executed that will control the spacecraft power switching to the capsule, or to cause quantitative command data to be sent to the capsule from the spacecraft Command Subsystem. This is essentially the same effect as proposed above for one of the uses of R-STARTS or C-STARTS.

   This interface can therefore be solved by giving the capsule STE essentially one of ACE's C-START's. Since there will probably be a considerable distance separating the STE's a data link (which could be similar to ACE's DTVC or as simple as the placing of 24 bits via tones on a telephone pair) will have to be provided. The spacecraft's STE end should contain a buffer and logic such that its operation looks like a C-START to the spacecraft ACE's CUE. The data to be sent to the capsule can be either a part of the C-START data or preprogrammed in the spacecraft STE's uplink computer memory; the particular mode to be employed at that instant can be a part of the C-START instruction bits. The execute function of the C-START would be performed by a verification signal originating from the capsule STC.

b. Downlink. The GE Task B telemetry format provided (depending on the telemetry mode) 112 or 16 or 0 seven-bit words that were to be obtained from the capsule and transmitted in a block at a rate of 17 words/second. These words must be made available to the capsule STE in real time. Existing ACE ground stations accomplish similar data outputting functions to peripheral equipment in blockhouses 34 and 37 and LCC 39. The method they use for sending event or status information (i. e., transmitting tones representing a 1 or 0 and combining up to 24 on a pair of telephone lines) can be used to send this seven-bit digital data, with up to nine bits of identification (7 minimum required) supplied by the spacecraft's decommutator and one bit for synchronization. By detecting tones, etc., at the capsule STE, the data can be recovered and processed as they desire.

4.2.4.4 DSN Interface

Figures 4-2 and 4-3 indicate that the STE is to be tied into the Deep Space Network DSS 71's command and telemetry links. This tie-in can be made at the spacecraft interface equipment and should therefore not perturb the existing ACE's equipment. The use of this link is to verify the compatability of the flight spacecraft with the DSN. As such, some controlling (via spacecraft command) and monitoring (via spacecraft telemetry) will be accomplished from the DSN. This control of the spacecraft will cause local evaluation problems if done randomly, and therefore should be accomplished with a fairly rigid agreed-upon sequence of commands and timing. This is a procedural problem and not a hardware problem.

4.2.4.5 Pad Operations

Figure 4-3 is the simplified block diagram of an ACE launch pad configuration; Appendix C contains a summary of the requirements. The significant points are:

a. There is a large reduction in monitoring requirements for pad operation as compared to system test.

b. Two spacecraft/capsule configurations require support at the pad.

c. Data (display) link to the LCC is required.

d. DSS 71 is on-line and is required to be in RF lock with the spacecrafts at time of launch.

e. Hardwire control and monitoring of the critical functions are available in the LCC.

f. Radio Subsystem SIE located in the STE is used to perform RF operations and is tied into a separate ACE uplink and downlink path so as to be able to act as backup for the normal data link. (These same points are to be tied into the DSS 71 complex.)

Item (a) has no effect on ACE; items (b) and (c) are identical to the Apollo requirements and thus do not perturb the ACE complex; item (d) (similar to the discussion in the previous section) also has no effect on ACE.

The hardwire controls and monitors (item e) should not cause any great concern as these can be introduced into the spacecraft interface equipment. By designing these controls and monitors into that equipment, the resulting affect on ACE can be minimized.

The effect of having the Radio Subsystem SIE in the STE, item (f), should not cause any uplink problems, other than requiring a separate DTVC path (the same as used during system tests) to be on-line during the launch operations. The recovery and decommutation of the spacecraft telemetry data obtained via the Radio Subsystem SIE as a backup for the normal data link will be capable of being patched into the Radiation decommutator where it will be handled by a special decommutation program. In this mode the umbilical data and the OSE data will not be available in the control room but only in the LCC via the hardwire circuits. Since this is a backup mode, this is satisfactory. It should be noted that the OSE data obtained by the Radio Subsystem SIE will not be available to the control and console group since that data is not being gathered.

It should also be noted that 10-bit OSE data (SIE and umbilical) is not required at the launch pad and therefore the existing ACE DTMS or a very simplified version of that equipment could be used. Since the STE DDAS is not being used at this time within the STE, the current approach is that it should be moved with the spacecraft as it moves through the KSC flow in order to maintain the same spacecraft-to-OSE interface.

4.2.5 FACILITY REQUIREMENTS

4.2.5.1 Factory Requirements

Appendix A, Apollo ACE Summary Description, contains a summary of the facilities required to support one Apollo ACE ground station. It is expected that the differences in Voyager requirements would not affect this noticeably. The differences identified to date are:

a. Voyager requires approximately ten less lowboy consoles in the control room, but this should be offset by the Science processing equipment required to decode the error-correcting data and to reconstruct their TV pictures, etc.

b. Voyager requires only one Radiation decommutator in the computer group. The removal of one decommutator from the computer complex will not appreciably alter the size, power, or air conditioning load.

The spacecraft interface equipment will consist of approximately 30 racks of electronic equipment that will require facilities similar to the ground station. These will include raised flooring, power (estimated as 75,000 watts of 120/208 volt power), air conditioning (estimated as 100,000 Btu/hr), and lighting. This equipment must be within 50 to 100 feet of the spacecraft but need not be in the clean room, thereby permitting more ready personnel accessibility to the SIE. Several pieces of high pressure gas equipment, occupying approximately 100 square feet, will also be required. The SIE will move with the spacecraft as it goes from the systems test area to the thermal vacuum area.

The immediate spacecraft-vicinity area will require approximately 3000 square feet of floor space per spacecraft. This area must be serviced by an overhead crane having a hook height of approximately 38 feet. The room should be a 100,000 class clean room. This area will contain the following major pieces of equipment:

a. Spacecraft.
b. Work stand.
c. Spacecraft cooling/heating equipment.
d. Sensor stimulation equipment.
e. Access stands.
f. Bay handling equipment.

This area and the SIE will be interconnected by 400 to 500 circuits, which will include several power leads, several coaxial lines and waveguides with the remainder being predominately twisted shielded pairs.

An extensive communications network between all areas, including television pick ups and displays, will be required.

4.2.5.2 KSC Requirements

The use of existing ACE stations at the Manned Spaceflight Operations Building (MSOB) requires that the control room (using the existing console) be disrupted only to the extent of providing enough room for the Science Subsystem data processing equipment. This can be accomplished by removing several lowboy consoles not required for other Voyager activities. If the human-factored consoles are used, all of the existing consoles must be removed. The new consoles will have as a design requirement that they be compatible with the existing facilities, including connector compatibility, air duct compatibility, power, etc., as well as the electronic interface compatibility.

The MSOB spacecraft-vicinity equipment and the spacecraft area will be as in the factory. This area must be able to support three simultaneous vehicle tests.

When the spacecraft is moved to the hazardous-material-handling area, the amount of spacecraft interface equipment can be reduced to approximately 15 cabinets. The volume required by the spacecraft and service stand would not be affected.

When the spacecrafts are moved to the launch pad, each will require approximately eight cabinets of SIE.

4.3 ACE SOFTWARE SYSTEM

4.3.1 GENERAL

The ACE hardware configuration provides uplink and downlink paths for command and response communication with the spacecraft. Commands are generated and transmitted to the spacecraft by the uplink computer as a result of information entered through the data entry system. Continuous monitoring of the spacecraft provides telemetry responses which are received through the decommutator. The ACE ground station contains a data processing system that consists of two CDC 160G general purpose digital computers and associated peripheral equipment. It is these two computers, and therefore the software system residing therein, that control the command generation and spacecraft data monitoring.

The ACE software system consists of three primary programs. These are the supervisory executive program (SEP), the uplink control program, and the downlink control program.

The SEP exercises overall control of the system programs and loading. SEP loads the uplink program, the downlink program, and all parameters associated with them. The uplink and downlink control programs provide the necessary subroutine to generate commands which are sent to the spacecraft and to process the digital responses generated in the spacecraft. Each of these programs consists of a number of subroutines that control specific functions to be performed. The system is shown in block diagram form in Figure 4-5.

4.3.2 PROGRAM FUNCTIONS

4.3.2.1 Supervisory Executive Functions

The supervisory executive program exercises overall control of the ACE system and its programs. In addition, the SEP locates the proper file on the test tape and loads the uplink and downlink control programs, uplink and downlink parameters, scope annotations, and calibration data. A specified C-START may initiate three different types of changes in the course of operations while this program is running. The changes that may be initiated by the C-START are:

a. Load an entire test.

b. Load a new annotation into SGS (symbol generation and storage) memory and load its associated downlink parameters.

c. Change the engineering units in a specific downlink parameter.

Also, there is a typewriter routine available that may be called up by either computer.

4.3.2.2 Uplink Control Functions

The uplink control program consists of two major parts: nonpriority and priority. The nonpriority part, the executive subroutine, maintains a logical flow of the noncritical processing tasks such as: initialization, computer-to-computer data transfers, and realtime maintenance checking. The nonpriority portion of this program may be interrupted at any

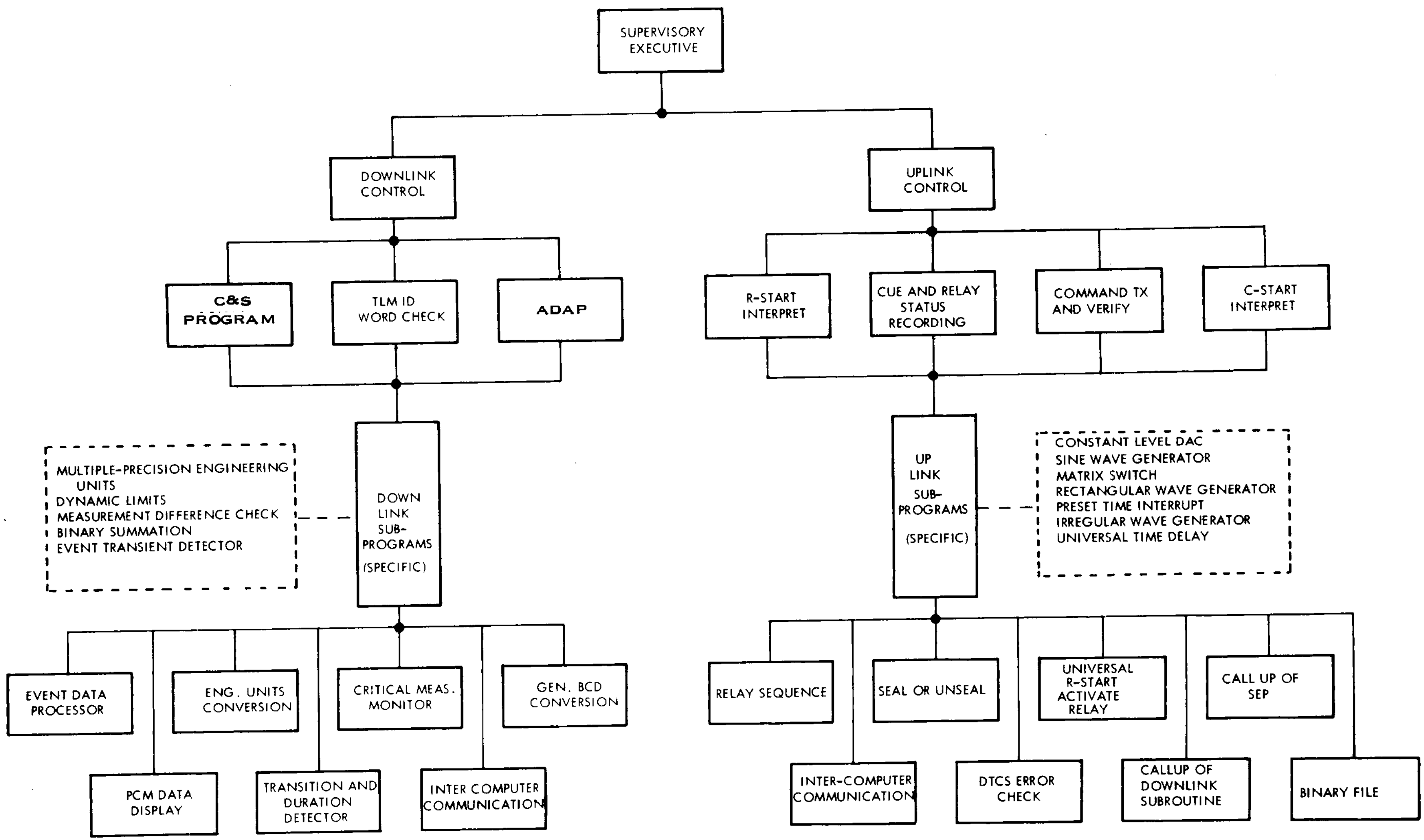


Figure 4-5. ACE-S/C Software System, Voyager Application

time to service the priority subroutine. The priority part consists of all the interrupt-handling subroutines and the subroutines directly affected by the execution of an interrupt. The various interrupts are the key to the program. By processing them, the program can determine the necessary action required to accommodate a test operation request.

4.3.2.3 Downlink Control Functions

The downlink control program is designed to process incoming telemetry data. This processing provides orderly completion of all required steps and exercises control from initialization of inputs through output to display and peripheral equipment. The functions of this program are performed through varous subroutines. These functions have four main groupings:

a. Primary Initialization. This function assures that registers, counters, and other indexes are reset to the start condition. Inputs are initiated by simulating a data interrupt.

b. Subroutine Linkage. The subroutine linkages consist of internal subroutines that control entry to and return from particular external subroutines. These internal subroutines provide the first parameter address for the data word being processed. They also determine if more data words are to be processed, and repeat the procedure if necessary.

c. Summary Filing. This function transmits data, as specified, from a common output area in memory to magnetic tape.

d. Self-checks. This function is performed on major I/O equipment to verify operation and indicate the downlink system status.

4.3.3 ACE AUTOMATION

The preceding describes the software system as it was first configured. In the Apollo program, this is known as the Block I configuration. It should be emphasized that for this configuration the majority of data entry and response interpretation is performed by control room operator personnel. After substantial ACE operating experience was gained, an adaptive intercommunication system was developed wherein many command operations have become automatic. That is, programmed, predetermined operations are executed by the computer. Thus, complete command generation, transmission and monitoring of the appropriate downlink response can be performed automatically by the CDC 160G computers. Although the adaptive intercommunication requirement is of subprogram nature, the specification in essence stipulates a "system" for spacecraft testing which uses the ACE computers in a closed-loop fashion. This new system is commonly referred to as Block II or ADAP.

4.3.4 APPLICABILITY OF ACE SOFTWARE TO VOYAGER

For the purpose of investigating the applicability of ACE software to the Voyager program requirements, the ACE software can be divided into two groups. The first group contains all of the general purpose programs generated as part of the Block I system and used primarily for manual testing, as well as all of the programs used to generate and control the execution of the specific adaptive intercommunications routines. The second group consists solely of the specific adaptive intercommunication routines used to control the specific parts of individual spacecraft tests. The first group will be referred to as system programs while the second group will be called ADAP routines.

The systems programs are directly applicable to the Voyager to the extent that the corresponding ACE hardware is used in the Voyager system without change. For example, the R-START interrupt routine of the uplink control program can be used without change. On the other hand, the programs associated with hardware that is changed in the Voyager application of ACE must be modified or replaced to reflect the hardware changes. A typical example of this is telemetry decommutation.

The telemetry system proposed for the Voyager spacecraft and the ground telemetry associated with the OSE may be compatible with the ACE decommutator; however, previous versions of the spacecraft (Task B for example) have used telemetry systems which were basically incompatible with the decommutator. With the Task B telemetry system, it is possible to decommutate the data in the downlink computer. This naturally would result in a significant software change to provide the necessary decommutator program and integrate it into the downlink computer program.

A preliminary estimate based on the spacecraft and OSE configuration developed during Task D indicates that a high percentage (approximately 90%) of this group of ACE programs can be used without change in support of Voyager testing. The remainder of the programs would require change; however, in most cases the new program could be generated by modifying existing programs.

One such modification is due to the method used to send commands to the spacecraft Command Subsystem. In Apollo, the K-START is used to enter data, which is processed by a subprogram and sent to the spacecraft by way of the DTVC and the G&N buffer in the DTCS. In the case of Voyager, the data would be entered through an R-START or a C-START and processed by the computer which would send it to the spacecraft by way of the DTVC, the G&N buffer in the DTCS, and the Command Subsystem portion of the spacecraft interface equipment. The only new item in the chain (other than the SIE) is the program which generates the DTVC signal based on the input from the start. This can be easily generated from the existing programs used with the K-START module.

Generally speaking, the ADAP routines will not be usable on the Voyager program. This incompatibility is due to the nature of ADAP routines. They are designed and written to accomplish a particular test on a specific spacecraft, and the occasion will be rare when an ADAP routine written for one spacecraft will find application on a different spacecraft.

Additionally, in order to test a specific spacecraft with the ACE system, the software system must be configured to reflect the specific spacecraft configuration and the specific test to be performed. This is accomplished by generating a test file tape which contains parametric data relative to the specific spacecraft and the subprograms required to perform the specific test. Programs are available which provide the capability of semiautomatic preparation of the test file tape for spacecraft testing. This test file tape is then loaded into the ACE computers and the system is ready to test the spacecraft in the desired fashion. Naturally, none of the existing test file tapes are applicable to Voyager.

SECTION 5

APPLICABILITY OF APOLLO/SATURN MECHANICAL AND FLUID GROUND EQUIPMENT

5.1 APPROACH

5.1.1 BASELINE REQUIREMENTS

Following establishment of the Voyager Spacecraft Configuration and Baseline Test Flow Plans (described in Section 3 of this report), an initial study was conducted to determine the general applicability of Apollo/Saturn Mechanical and Fluid Ground Equipment to Voyager.

In this study, it became apparent that the Saturn launch vehicle fluid service equipment was not appropriate for Voyager application because: (1) the equipment was sized to handle flows far in excess of Voyager requirements; (2) the Saturn launch vehicle fluid servicing equipment handles RP-1 and cryogens, while the Voyager uses hypergols; and (3) servicing of the Voyager spacecraft (like Apollo) would be accomplished via the mobile service structure (MSS), with feed systems and access at the required levels, while the Saturn GSE services the launch vehicle through the launch umbilical tower (LUT).

Therefore, with the additional consideration that interchange of Apollo and Voyager spacecraft payloads might make Apollo spacecraft GSE readily available to Voyager, it was decided to concentrate on the Apollo spacecraft system GSE--specifically that supporting the command module, service module, and the lunar excursion module.

5.1.2 EVALUATION CRITERIA

Two consecutive criteria were applied to evaluate potential applicability of Apollo GSE to Voyager:

> First: "Can the equipment reasonably be used (or modified for use) on Voyager?" (A further presentation of technical considerations involved in this analysis is included as Section 5.2.)

Second: "Would it cost more to modify than to design and build from scratch?"

5.1.3 EQUIPMENT IDENTIFICATION

Two documents published by Apollo prime contractors were utilized as the primary source to determine available Apollo GSE. These were: North American Aviation Spacecraft GSE Index, and Grumman Handling and Transportation GSE Pictorial Index. (These documents listed information on approximately 1000 items of GSE.)

A review of these initially resulted in the selection of approximately 150 candidates for applicability to Voyager. To obtain more detailed information on certain of these items, NAA Interface Control Documents and GSE End Item Specifications, and Grumman GSE Summary Data Sheets were obtained.

Information in these documents, supplemented by discussions, where applicable, with GE-MSD personnel working on this task who have several years experience with the Apollo/Saturn program at KSC, formed the basis for the final selection of the 95 items which are recommended for application to Voyager. (These are identified in Section 5.3 and detailed in Appendix D.)

5.2 EVALUATION CRITERIA

The evaluation criteria used for the initial equipment review was simply: "Can the equipment reasonably be used (or modified for use) on Voyager?"

More specifically, this involved evaluation of each type of equipment item with respect to the following factors:

a. Propellant Handling Equipment

   1. Fuel or oxidizer used. (Problems of compatibility of Voyager physical and chemical properties and their effect on the Apollo equipment. What changes in materials, seals, and methods of operation might be needed?)

2. Condition of fluids delivered by the equipment. (Temperature, pressure, and delivery flow capability of the Apollo equipment compared to Voyager requirements. This included an increase or decrease in fluid conditioning time, changes in size of storage tanks, etc.)

3. Location and accessibility. (Proximity of the equipment to the Voyager spacecraft; location on the pad; location of the equipment on the Mobile Service Structure or elsewhere on the pad.)

4. The practicability of modification. (Was it feasible to use the equipment without modification? Could an adapter be introduced into the system? If modification was required, what was the extent of modification required?)

5. Effect on Voyager plans. (What were the operating requirements on the Apollo equipment that were different from those initially planned for Voyager? Assuming that the equipment and method of use were not changed, could the Voyager procedures be changed without major effect on operational plans?)

b. Gas Handling Equipment

1. Type of gas available. (Voyager requirements are for nitrogen gas for attitude control, and helium gas for propulsion system pressurant. Would the Apollo gas service equipment have the required capability?)

2. The amount (cubic feet) available and the temperature and pressure conditions at which it is available.

c. Spacecraft Checkout Equipment. Since this category included those items of bench checkout or hand-carried type of equipment, and is basically general purpose equipment used for particular applications in the checkout procedures, the evaluation considered the similarity of function performed by the equipment, and the probability of much improved equipment being available from commercial sources at the time needed for Voyager. It did not dwell on the full extent of the method of use and point of use in the overall test and checkout schedule. These factors were weighed against the probability of the aging Apollo equipment still being available several years from now.

d. Spacecraft Handling Equipment

1. Slings. Since the LEM is of the same general size and weight, slings for LEM were appraised for similarity in weight lifting capability, similarity of items of equipment being lifted--size, shape--and the feasibility of adapting the LEM slings and the attachment hardware.

2. Dollies. In addition to the factors considered for slings, consideration was given to the degrees of freedom of positioning, types of adapters required and the feasibility of providing them, and the extent of modification required.

e. Transportation Equipment/Shipping Containers

1. Size, configuration.

2. Compatibility with proposed shipping methods.

3. Environment control.

4. Feasibility of modification.

The second evaluation review was made with additional technical data being available in many cases, and with the added criteria of: "Would it cost more to modify than to design and build from scratch?" Availability of equipment from the Apollo program for modification or adaptation for Voyager use, and the effect of age, being out of date, and equipment wear-out were also considered.

5.3 RESULTS

5.3.1 GENERAL

The first review of the two index documents containing approximately 1000 items resulted in the selection of approximately 150 items for further consideration. The second review resulted in the selection of 95 items which met both the "feasibility" and "practicability" criteria. (Data sheet summaries for each item selected are presented in Appendix D. These summaries briefly describe the equipment, its function, and its applicability to Voyager.)

The Apollo equipment which can provide the greatest benefit to the Voyager program is the fluid service, or propellant servicing equipment. The propellant servicing equipment for the command module, service module and lunar excursion module were evaluated. Since the LEM descent engine is used on the Apollo program and is proposed for the Voyager program, it followed that the LEMDE propellant servicing equipment should be usable; this is the case and relatively little change or adaptation is required to utilize this equipment on Voyager. The other propellant servicing systems were less desirable for either of two reasons: either the capacity is inadequate or the fluids provided are not compatible with Voyager requirements.

In some instances, where the equipment capability is less than Voyager requirements but the fluids were compatible and modification would be possible, the equipment was included and should be considered as a backup. Review of those items later in the program is recommended.

It was also determined that the LEM most nearly resembled the Voyager Spacecraft in size and weight, so that the LEM handling equipment could be utilized with the least modification being required. The results of the study, for the five areas of equipment evaluated, are presented in the following paragraphs. Data sheets on each item of equipment selected are presented in Appendix D. and are grouped according to the following paragraphs.

5.3.2 PROPELLANT HANDLING EQUIPMENT

The LEM descent engine propellant servicing equipment is considered applicable for use on the Voyager program, especially if the LEMDE is used for Voyager. The LEMDE uses Aerozine 50 and N_2O_4 as propellants. Voyager will use N_2O_4 as the oxidizer, and either Aerozine 50 or MMH as the fuel. The two fuels are, for purposes of equipment design, identical. The use of either in the LEMDE fluid service equipment should not present a problem for compatibility, methods of handling, seals, and valving. The method of disposal of toxic vapors is applicable for either fuel. The principal advantage, in addition to proper sizing, is the location of the LEM equipment on the mobile service structure. It is significant that all the fluid service lines, inert gas lines, valve control boxes, controls and remote tie-in capability for remote operation, exists on the MSS. Thus, a significant benefit can result from the LEM fluid service equipment utilization. Figure 5-1 is a block diagram of the significant identifiable equipments used to service the Apollo spacecraft system, with those items applicable for Voyager indicated by crosshatching.

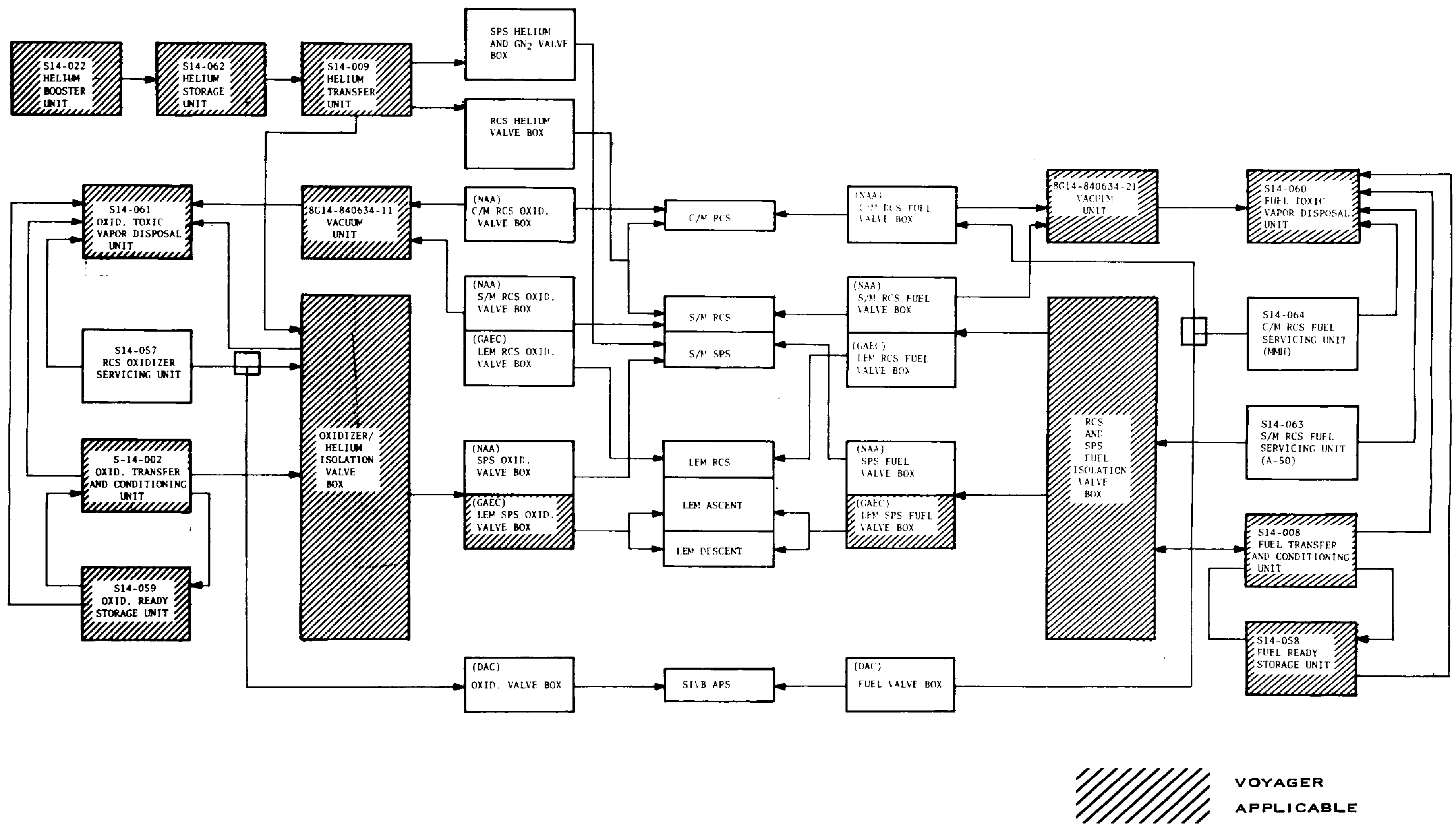


Figure 5-1. Apollo Fluid Servicing Equipment, Block Diagram

Data sheets for these selected items are included in Section 1 of Appendix D:

Fuel

SPS Fuel Bleed Measuring Equipment, S14-124-101 (NAA)

Fuel Transfer and Conditioning Unit, S14-008-201 (NAA), LSC-430-94008 (GAEC)

Fuel Ready Storage Unit, S14-058-0001 (NAA), LSC-430-94058 (GAEC)

Fuel Vapor Disposal Unit, S14-060 (NAA), LSC-430-94060 (GAEC)

Apollo S/M RCS Fuel Servicing Unit, S14-063 (NAA), LSC-430-94063 (GAEC)

Apollo C/M RCS Fuel Servicing Unit, S14-064 (NAA)

Weigh Tank Calibration Unit, LDW 430-6190-1 (GAEC)

Fuel Valve and Control Boxes (NAA)

Oxidizer

Oxidizer Transfer and Conditioning Unit, S14-002-201 (NAA) LSC-430-94002 (GAEC)

Apollo RCS Oxidizer Servicing Unit, S14-057 (NAA), LSC-430-94057 (GAEC)

Oxidizer Vapor Disposal Unit, S14-061 (NAA), LSC-430-94061 (GAEC)

SPS Oxidizer Bleed Measuring Equipment, S14-122 (NAA)

Oxidizer Valve and Control Boxes (NAA)

5.3.3 GAS HANDLING EQUIPMENT

Two gases are proposed for Voyager. Helium will be used as the pressurant for the propellant tanks, and nitrogen is planned for use as the attitude control propellant. Nitrogen gas is not used on the Apollo; therefore, no GSE exists for that purpose. Nitrogen being used comes from the facility that has more than adequate capability of providing gas at the required pressure and in sufficient volume for Voyager requirements. Servicing equipment for the Voyager attitude control system will be required. Helium gas is used on Apollo for the

LEMDE propellant tanks pressurization and is of adequate capacity for Voyager. Interface connections at the Voyager spacecraft are not considered to be a significant problem.

Data sheets for these selected items are included in Section 2 of Appendix D:

Helium

Helium Transfer and Conditioning Unit, S14-009 (NAA), LSC-430-94009 (GAEC)

Helium Booster Unit, S14-022 (NAA), LSC-430-94033 (GAEC)

Helium Ready Storage Container, S14-062-0001 (NAA), LSC-430-94062 (GAEC)

Helium Valve and Control Boxes (NAA)

Nitrogen

Protective Pressurization Unit Assembly, S14-099 (NAA)

5.3.4 SPACECRAFT CHECKOUT EQUIPMENT

The specialized spacecraft checkout equipment used on the Apollo spacecraft system has, in several instances, a function which is similar to that required for Voyager. Alignment sets and accessory equipment such as angular alignment, linear measuring accessories, leveling instruments, and target equipment, all perform similar functions. Until hardware design is nearing completion, the final method or technique of performing these functions cannot be firmly determined. Therefore, the evaluation took into account the fact that similar functions would be required, and if the Apollo equipment item performed a function similar to that required for Voyager, and was of acceptable size, shape, and probably would not be significantly outdated by the development of new and more accurate instruments, it was identified as being applicable for Voyager. Final design, and review of the spacecraft (Voyager) requirements, may change this evaluation.

General purpose instruments of the general laboratory bench type were usually not considered because the latest and most applicable equipment surely should be available from the contractors' laboratory capability.

Data sheets for these selected items are included in Section 3 of Appendix D:

Spacecraft Test

Helium Leak Tester Mass Spectrometer, S14-003 (NAA)

Propulsion System Fluid Checkout Unit Console Assembly, C14-075-301 (NAA)

Adapter Unit - PUGS SPS, C14-352 (NAA)

Descent Engine Simulator Live Propellant, LDW-430-6150 (GAEC)

Thrust Chamber Assembly Alignment Equipment, C14-408-0001 (NAA)

Pyrotechnic Initiators Substitute Unit, A14-003 (NAA)

Pyrotechnic Initiators Substitute Unit, A14-139 (NAA)

Optical Alignment Set, A14-028 (NAA)

Electronic Weighing Kit, H14-041 (NAA)

Box Level, A14-047 (NAA)

AOT Optical Target Set, LDW 420-13371 (GAEC)

Descent Engine Plug Assembly, 420-63420 (GAEC)

Descent Engine Leak Test Set, 420-62366 (GAEC)

Mobile Optical Alignment Equipment, 420-13360 (GAEC)

Auxiliary Test

Engine Decontamination Unit, S14-070 (NAA), LSC 430-94070 (GAEC)

RCS Freon Flush Cart, LDW 430-6860 (GAEC)

RCS Engine and Purge Unit - Oxidizer, LDW 430-2490 (GAEC)

Flowmeter Cart, 30-60 scfm, 430-6420 (GAEC)

Calibration Unit - Pressure, 6000 psig, C14-426 (NAA)

Flow Rate Calibration Set, C14-427 (NAA)

Temperature Calibration Set, C14-428 (NAA)

Ground Cooling Cart, A14-011 (NAA)

Ground Air Circulating Unit, A14-036 (NAA)

Vacuum Cleaner, A14-035-0002 (NAA)

Temperature Controlled Battery Storage Rack, 420-83280 (GAEC)

Test Fixtures and Stands

C/M Optical Alignment Support Equipment, A14-135 (NAA)

Base Support Stand, H14-031 (NAA)

Command Module Maintenance Stand, H14-035 (NAA)

Access Stand for External S/C, H14-109-101 (NAA)

Spacecraft Integrated Systems Workstands, H14-124 (NAA)

S/M and S/C Adapter Weight and Balance Fixture, H14-9059 (NAA)

Weight and Balance Jack Set, H14-9015-101 (NAA)

Weight and Balance Fixture, LDW 420-13460 (GAEC)

Descent Engine Support Fixture, 420-6043 (GAEC)

Descent Stage Propellant Tanks Installation Fixture, LSC 420-63150 (GAEC)

LEM Turnover and Handling Fixture, LDW 420-13110 (GAEC)

Three Axis Positioner Work Platform, 420-13220 (GAEC)

LEM Integrated Workstand, 420-13390 (GAEC)

Cleaning Positioner, LDW 420-13750 (GAEC)

Polarity Checker, LSC-420-93089 (GAEC), G14-818100 (NAA)

Polarity Checker Work Platform, 420-31040 (GAEC)

Optical Alignment Fixture, LDW 420-13360 (GAEC)

5.3.5 SPACECRAFT HANDLING EQUIPMENT

As the handling equipment and slings used for the LEM and its components are of similar size and weight to those required for Voyager, they were the primary source of candidate equipment. Until the detailed designs of the Voyager spacecraft are completed, the final decisions as to the method of handling, and in particular the designs of the handling slings, lifting devices, position devices, and installation devices, can only be generalized. Therefore, the equipment recommended in this section is principally for the descent engine (LEMDE) and general purpose handling equipment, such as auxiliary crane controls.

Data sheets for these selected items are included in Section 4 of Appendix D:

Spacecraft Handling Equipment

Auxiliary Crane Control, A14-046 (NAA)

Auxiliary Crane Control, A14-134 (NAA)

Auxiliary Crane Control, 420-13060 (GAEC)

Mobile Crane, 420-1290 (GAEC)

Portable Winch Assembly, LDW 420-13311 (GAEC)

S/M Fuel and Oxidizer Tank Sling, H14-060 (NAA)

Spacecraft Sling (without LES), H14-073 (NAA)

Fuel and Oxidizer Tank Sling, H14-102 (NAA)

Spacecraft Sling, CSM, SLA, LEM, H14-212 (NAA)

Descent Engine Installation Sling, 420-63500 (GAEC)

Descent Engine Turnover Sling, LSC 420-63511 (GAEC)

LEM Inverted Hoisting Sling, LDW 420-13068 (GAEC)

LEM Vehicle Hoisting Sling, 420-1300 (GAEC)

Battery Handling Mobile Hoist Rig, 420-83109 (GAEC)

Descent Stage Handling Dolly, 420-13550 (GAEC)

Descent Stage Propellant Tank Dolly, LDW 420-63980 (GAEC)

Descent Stage Engine Installation Dolly, 420-63400 (GAEC)

SIVB LEM Adapter Base Assembly, H14-165-101 (NAA)

SIVB LEM Adapter Base Assembly, H14-166-101 (NAA)

5.3.6 TRANSPORTATION EQUIPMENT

The transportation devices used for the LEM are of the same general size and weight handling capability as required for Voyager. There were specialized transporting devices that were considered not to be feasible to attempt to modify for Voyager. The primary consideration was the fact that the Voyager spacecraft will be shipped to KSC and handled there as a complete vehicle. The transportation container must be compatible with the proposed methods of shipment, which are by air in the Super Guppy, and by barge. The former places a limitation on the external size and configuration of the container. None of the Apollo system shipping containers met the combined requirements of fitting in the Super Guppy and being able to contain the assembled Voyager spacecraft. However, various tow trucks and trailers used on Apollo can be used to transport the Voyager at a test complex, or within a building.

Data sheets for these selected items are included in Section 5 of Appendix D:

Spacecraft Vertical Transport Vehicle, H14-131 (NAA)
Base Closure, SIVB LEM Adapter, A14-157 (NAA)
Wheeled Warehouse Tractor, Gas Powered, 420-13330 (GAEC)

5.4 FURTHER STUDY

The 95 items of equipment described on the data sheets in Appendix D are applicable for use on the Voyager Program. As has been stated, the items selected may require varying amounts of modification or the use of adapting "black boxes," but the use of the equipment recommended will significantly benefit the Voyager Program. Several areas which should have further study are presented and discussed below.

5.4.1 CONSIDERATION OF COST

Cost was considered in the equipment evaluation, as a matter of engineering judgment. There is a point in the modification of existing equipment where it is best to incur a slight increased cost and have the benefit of new equipment which meets the requirements exactly. It is recommended that the equipment selected be subjected to preliminary designs of the modifications required, so that this assessment can be made.

5.4.2 EQUIPMENT AVAILABILITY

The evaluation considered only whether or not the Apollo equipment function and capability were applicable to Voyager. Consideration was not given to the total number of each item of GSE on the Apollo Program, its physical location away from KSC, or the time requirements of availability for modification in preparation for use on Voyager. Also, the possible continuing requirements for the use of the Apollo equipment on Apollo follow-on programs should be assessed and an overall schedule of utilization determined.

5.4.3 EQUIPMENT LIFE

The procurement specifications for equipment having rotating parts and other components subject to wear, typically specified a life ranging from three to five years. With preventative maintenance and renovation, a total use life approaching ten years can be predicted. However, the equipment wear-out factor should be evaluated thoroughly in view of the long life span required for use of the equipment for Voyager.

SECTION 6

REFERENCES

The following reference documents were utilized during the Study:

Voyager Spacecraft System, Phase 1A Task B Preliminary Design (DIN 655D4514)

North American Aviation Spacecraft GSE Index, Document SM-3A200

Grumman Handling and Transportation GSE Pictorial Index

North American Aviation GSE End Item Specifications

North American Aviation Interface Control Documents

Grumman GSE Summary Data Sheets

Description Manual ACE S/C, NASW-AM03

Description Manual MILA Peripheral Equipment LC 39, NASW 410 AM-63

Programming Standards Manual ACE S/C, ACE S/C Program Office, MSC Florida Operations.

APPENDIX A

APOLLO ACE SUMMARY DESCRIPTION

A.1 GENERAL

The ground station of an Apollo ACE is shown in simplified block diagram form in Figure A-1. The system can be considered to have three functional groupings, i.e., the command (uplink) equipment, the measurement (downlink) equipment, and the spacecraft interface equipment (SIE). These groups are described briefly below.

A.2 COMMAND LINK

A.2.1 START EQUIPMENT

The system test of each spacecraft subsystem is controlled from START (selections to actuate random testing) modules located on console equipment in the control room area. START commands, after computer processing, are transmitted to the digital test command system (DTCS) in the vicinity of the spacecraft for decoding and issuance to the SIE.

The START modules are located in varying numbers on the system control consoles. There are three distinct types of START modules: relay selection START (R-START) modules, providing for manual control of discrete events; computer communication START (C-START) modules, providing for manual selection of computer subroutines and the parameters required by the subroutines; and a single keyboard START (K-START) module, providing for manual and automatic insertion of information into the Guidance and Navigation Subsystem of the spacecraft.

A.2.2 R-START MODULE

The R-START module contains four pushbutton function switches, an execute (XEQ) pushbutton, and appropriate lights to indicate the status of the switches and of the module. The function switches provide the means to select the specific discrete events desired, and the XEQ switch initiates the sequence resulting in their occurrence. Each module controls the occurrence of

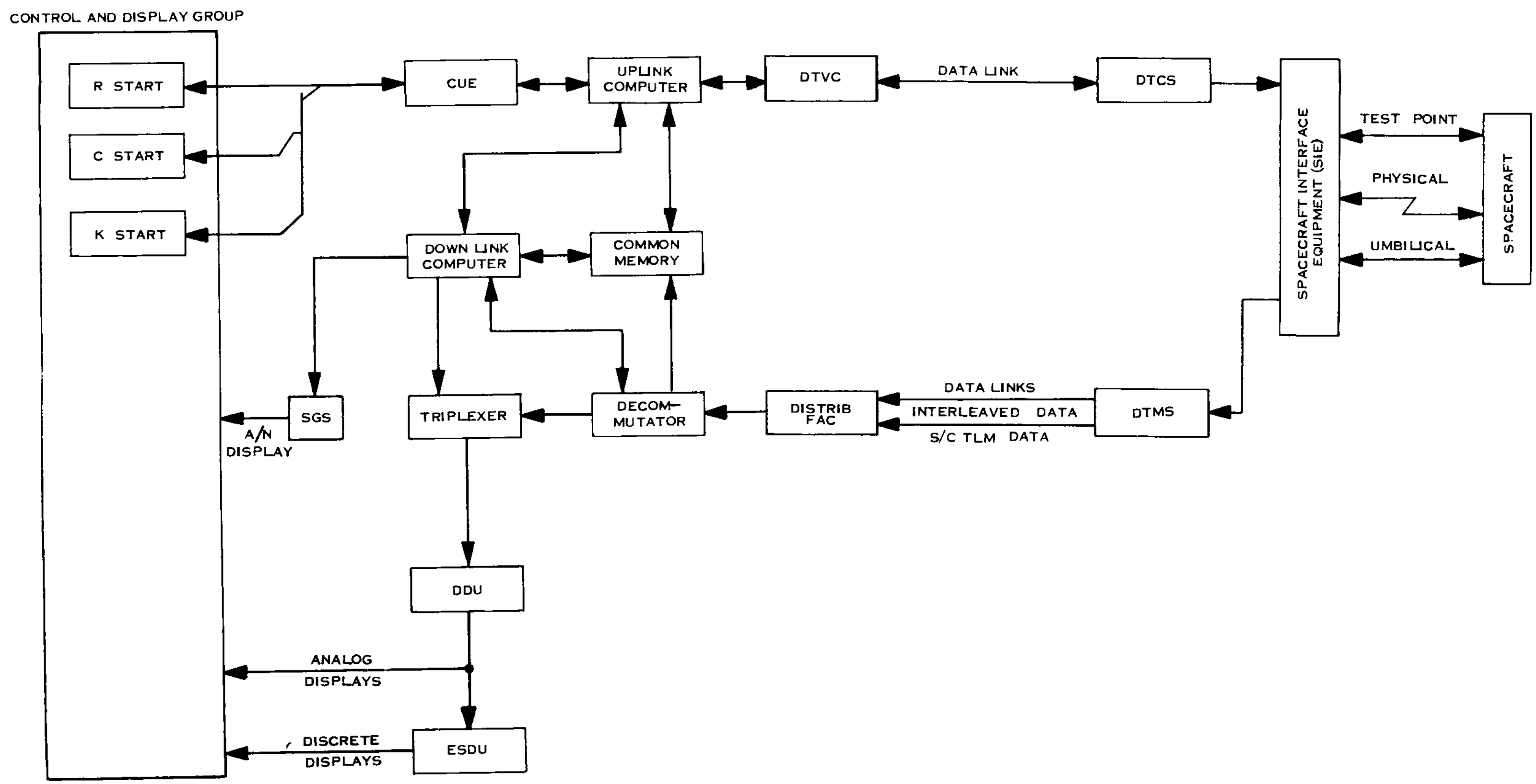


Figure A-1. Apollo ACE, Simplified Block Diagram four discrete events (e.g., individual relay action at the spacecraft vicinity) by providing a logical "1" or "0" from each function switch. The presence of a "1" is established by the on or off condition of the function switch.

A.2.3 C-START MODULE

The C-START module includes ten 12-position switches that select a specific command function. Each individual switch position provides a 4-bit digital word in binary-coded-decimal form for transmission to the computer. Thus, the output of the C-START module is a 40-bit command message and an 8-bit address that instructs the computer to perform specific operations (and also may provide parameters for these operations) and instructs the computer in the disposition of the results. The C-START panel also includes status indication of the switches, an XEQ pushbutton to initiate the transmission, and appropriate lights to indicate the module status.

A.2.4 K-START MODULE

The K-START module may be operated either manually (using a keyboard)or automatically (using a perforated tape reader). The keyboard provides 18 pushbutton switches. Depressing any one of these switches initiates the transmission of a binary word to the Guidance and Navigation Subsystem. The module panel also includes tape reader control switches. These switches provide for automatic sequencing of binary words from the tape or manual sequencing including both forward and reverse steps. A visual tape character-readout display is provided. Appropriate display lights on the panel indicate the status of the module.

A.2.5 J-BOX

J-boxes are located at various test consoles and allow all START modules to be connected to a common data and control bus. Sixteen J-boxes are used in the ACE ground station. Each J-box supplies power to the START module enclosures connected to it. The capability of these power supplies limits the number of START enclosures connected to each J-box to four. A START enclosure may contain 1 C-START or 4 R-START modules, or a maximum of 16 R-START or 4 C-START modules for each J-box.

A.2.6 COMMUNICATIONS UNIT EXECUTOR

The primary function of the communications unit executor (CUE) is to control the two-way communication path between the test operators and the command computer. To accomplish this control function, CUE sequentially interrogates all START modules in a repetitive fashion. The depression of an XEQ pushbutton (R-START or C-START) or a K-START keyboard pushbutton results in halting of CUE interrogation when it reaches that point in its scanning cycle. CUE accepts the command data and an address from the START module, compares this address with the scanner address, and transmits the data and the address to the computer. Data transmission to the computer is accomplished in two ways. If the data is from an R-START, the data and the address will be transferred as a single 12-bit word in parallel format. If the data is from a C-START or K-START, the data and the address will be transferred as four consecutive, 12-bit words in parallel format. When instructed by the computer, the CUE transmits verification replies to appropriate lights on the START modules.

A.2.7 COMMAND COMPUTER

The command computer receives the addressed data words from the CUE and performs the required processing under program control. The output of the computer consists of 24-bit messages that are applied to a data transmission and verification converter (DTVC). Each 24-bit message is a separate command intended for equipment in the digital test command system (DTCS) at the spacecraft test area. When the command computer receives verification of proper message delivery from the DTCS via the DTVC, the CUE is instructed to indicate verification on the appropriate START module panel. All input command data are recorded on a magnetic tape recorder.

A.2.8 DATA TRANSMISSION AND VERIFICATION CONVERTER

The DTVC is a two-way communicator and a parallel-to-serial and serial-to-parallel converter. All computer input/output (I/O) communications are in parallel format, but all transmissions over the lines to and from the receiver-decoder at the spacecraft test area are in serial format. The command computer input to the DTVC is a 24-bit message. This message during normal operation is transmitted twice to the DTVC. The DTVC reformats these two messages and transmits them to the receiver-decoder in the DTCS at the spacecraft test area. The first transmission of the message is stored by the receiver-decoder. The second transmission is compared with the first by the receiver-decoder. If the two transmitted messages compare, the receiver-decoder will execute the stored message.

A.2.9 DIGITAL TEST COMMAND SYSTEM (DTCS)

The basic function of the digital test command system is to receive serial-digital command data from a DTVC, decode the command data, convert the command data into stimuli and command signals, and route the commands and stimuli to the various systems under test. An additional function is to perform internal checks in decoding and selection to prevent double interrogation of equipment and to inform the ACE-S/C uplink computer of the status of the command transmission.

A single DTCS consists of a receiver-decoder, baseplates, relay modules, analog modules, and a guidance and navigation buffer unit. The function of the receiver-decoder is to receive, format, time, and check the redundant serial command from the ACE-S/C uplink data transmission equipment. The baseplates house the relay and/or analog modules, and the baseplate logic performs decoding for baseplate selection. The relay modules provide relay closure outputs to the SIE or service equipment. The analog module is basically a digital-to-analog converter that produces an analog stimulus in accordance with the uplink test command. The guidance and navigation buffer unit is used to generate serial digital data to the spacecraft or SIE. A single DTCS in its maximum configuration will contain 1 receiver-decoder, 32 baseplates, and 1 guidance and navigation buffer unit. The 32 baseplates are divided into 4 groups of 8 baseplates each. Each baseplate can contain 4 plug-in type modules. The plug-in modules provide system capability to be expanded or contracted to meet particular checkout requirements.

A.3 DOWNLINK

A.3.1 DIGITAL TEST MEASUREMENT SYSTEM (DTMS)

The purpose of the digital test measurement system (DTMS) is to acquire, condition, digitize, and interleave spacecraft and ground service equipment data and then to transmit this data over hard lines to the ACE-S/C ground station processing and display equipment. The DTMS is divided into four major equipment groups:

a. Airborne (A/B) response system.
b. Carry-on (C/O) response system.
c. Servicing equipment (S/E) response system.
d. Data-interleaving system.

Each response system contains commutating, encoding, and digital multiplexing equipment which is necessary to convert spacecraft and service equipment sensor outputs into a serial PCM signal. The three response system outputs (airborne, carry-on, and service equipment) are fed to an interleaver that combines the three separate PCM input signals into a single interleaved (I/L) serial PCM output at a rate of 204.8 kilobits per second. The A/B PCM signal, in addition to being interleaved with the other PCM inputs, is also transmitted intact (at its original rate of 51.2 kilobits per second) to the ACE-S/C ground station.

Each response system (airborne, carry-on, or service equipment) commutates the transducer output data at a rate of 50 prime frames per second and transmits the data at a rate of 51.2 kilobits per second with 128 words (analog or event) per prime frame. The output of each response system is applied to the interleaver and is commutated into a single serial bit train of 50 prime frames at 204.8 kilobits per second, with 512 words per prime frame. Thus, the interleaver commutates and transmits the PCM data to the ACE-S/C ground station at four times the input bit rate. The interleaver data is biphase modulated before transmission to the ACE-S/C ground station.

A. 3. 2 VIDEO DISTRIBUTION AND TAPE TRANSPORT CONTROL UNIT

The incoming spacecraft telemetry and interleaved data signals are routed via the terminal patch equipment to the video distribution equipment in the ACE computer room. This equipment provides suitable patching and switching functions to route the data to the desired decommutator or magnetic tape recorder.

A. 3. 3 PCM DECOMMUTATOR

Figure A-1 shows the PCM decommutator as processing either the spacecraft telemetry signal or the interleaved signal. In the interleaved mode, the spacecraft checkout and OSE data signal is decommutated in the normal manner. Parameter limit checks, floating limit checks (as required), and event transition checks are performed by the decommutator. After the checks are made, the data and the status of the checking are made available to the downlink computer external memory for storage in the location defined by the decommutator stored downlink computer external memory for storage in the location defined by the decommutator stored address. Each data word, with appropriate address, is also routed to the event and analog data distribution system for subsystem test console display.

A. 3. 4 DISPLAY COMPUTER

The display computer has access to the spacecraft checkout and OSE data which are stored in the 169G memory module through the direct access computer interface. Processing functions performed by the display computer include the following:

a. Conversion of selected data to engineering units and binary-coded octal words for transfer to the alphanumeric (A/N) symbol generation and storage (SGS) unit.

b. Transfer of selected spacecraft telemetry data and address to the event and analog data triplexer.

c. Transfer of out-of-limits and selected data, timing signal, and sync information to a magnetic tape recorder.

A. 3. 5 ALPHANUMERIC DISPLAY SYSTEM

The symbol generation and storage unit has the capability to store 480 lines of information with up to 40 characters in each line. The display computer initially transfers annotation data and parameter value data to the SGS unit and updates the parameter data as required. The number of parameters displayed on each line is dependent on the format desired. The data is displayed on cathode-ray tubes (CRT) mounted on the operator consoles. The operator monitoring the CRT has the capability of selecting any two "1/2 page" (12 lines) groupings of the formatted data. Summary indications of the other pages are given across the top and bottom. Out-of-limit values can be flagged by causing them (or their line) to blink.

A. 3. 6 EVENT AND ANALOG DATA TRIPLEXER

This unit accepts data inputs from the two decommutators and the display computer and provides a time-shared output of parallel data and address words to the decommutator distribution unit (DDU) for distribution of event and analog data.

A. 3. 7 DECOMMUTATOR DISTRIBUTION UNIT

The DDU receives addressed digital test data from the triplexer, distributes the test data to the event storage and distribution unit (ESDU) and to the test consoles for analog display. Analog words are identified by address recognition at the console and are stored by logic circuits contained within the console. The storage register outputs are fed to digital-to-analog (D/A) converters for routing to meter modules and analog recorders.

A. 3. 8 EVENT STORAGE AND DISTRIBUTION UNIT

Test data event words are identified by address recognition and routed to storage registers within the ESDU. The outputs of the storage registers are distributed to event indicators and event recorders at the test consoles.

A. 3. 9 CONSOLE DISPLAY AND RECORDING EQUIPMENT

Console display and recording equipment includes groupings in various numbers of the following:

a. Event modules each containing 24 indicator lamps.

b. Meter modules each containing 4 edge-mounted meter movements.

c. 8-channel analog recorders with a frequency response of approximately 75 cps.

d. 32-channel continuous write event recorders with a frequency response of about 100 cps.

e. CRT units.

f. The START modules are also mounted on these consoles.

A.4 SPACECRAFT INTERFACE EQUIPMENT (SIE)

The spacecraft interface equipment will be controllable from the DTCS to provide digital and analog inputs to the spacecraft and distribute the responses to the DTMS for transmission to the ACE ground station for display and processing. This equipment will interface with the spacecraft umbilical points and in special test configurations will use additional direct access test points. This equipment includes such items as ground power supplies, simulation equipment, RF equipment, etc.

A.5 FACILITY REQUIREMENTS

ACE-S/C ground station facility requirements are as follows:

a. Space Requirements

1. Control room	1500 (43 x 35)
2. Computer room	1450 (48 x 30)
3. Terminal facility room	700 (25 x 28)
4. Card preparation room	460 (20 x 23)
5. Maintenance room	500 (20 x 25)
6. Miscellaneous	400
Approximate Total	5000 square feet

b. Power Requirements

1. Shielded, filtered, facility ac power is required for the computer, control, and terminal facility rooms as follows:

(a) Voltage (nominal)	120/208
(b) Phase and connection	3 phase, 4 wire, wye connected
(c) Voltage tolerance and rate of change	±10 percent, 5.0 volts/sec
(d) Frequency (nominal)	60 cps
(e) Frequency tolerance and rate of change	±3 percent, 1.5 cps/sec
(f) Minimum power factor	0.8 lead or lag
(g) Demand factor	1.0 (unity)

2. Utility ac power is required for the card preparation room and maintenance area as follows:

(a) Voltage (nominal)	120/208
(b) Phase and connection	3 phase, 4 wire, wye connected
(c) Voltage tolerance	±10 percent
(d) Frequency (nominal)	60 cps
(e) Frequency tolerance	±4 percent
(f) Power factor	0.8 lead or lag
(g) Demand factor	1.0 (unity)

3. Estimated power requirements

	1 Phase	3 Phase
(a) Control room	1,400 watts	59,000 watts
(b) Computer room	56,000 watts	45,000 watts
(c) Terminal facility room	5,800 watts	9,700 watts

(d)	Card preparation room	8,500 watts	--
(e)	Maintenance area	2,500 watts	1,000 watts

4. Estimated total power requirements for one station:

   (a) 1 phase/120v/60Hz 75,200 watts

   (b) 3 phase/208v/60Hz 114,700 watts

c. Environmental Control

1. Total cooling requirements (estimated)

   (a) Computer room: Room-cooled equipment 32,290 Btu/hour; equipment cooling system 174,500 Btu/hour.

   (b) Control room: Total cooling load 138,280 Btu/hour.

   (c) Terminal facility room: Total cooling load 43,140 Btu/hour.

   (d) Card preparation room: Total cooling load 18,600 Btu/hour.

2. Equipment air conditioning accomplished by closed loop cooling system.

3. Cooling air supplied by ducts to base of racks in control room and exhausted to return ducts between racks.

4. Cooling air supplied by ducts to base of racks in computer and terminal facility rooms and exhausted through top of racks to return ducts. Several free-standing computer peripheral racks cooled by ambient comfort conditioning air.

5. All air supplied to racks filtered by mechanical filters at rack air inputs.

6. Temperature of equipment input air: 60°F dry bulb.

7. Nominal temperature rise through racks: 15°F.

8. Temperature of comfort conditioning air: 58° to 64°F wet bulb and 73° to 77°F dry bulb.

9. Relative humidity of equipment input air: 75 percent maximum in control room and racks with CRT's; 90 percent maximum in remaining areas.

10. Relative humidity of comfort conditioning air: 40 to 60 percent.

11. Equipment mounted on raised floor. Space beneath floor serves as return air plenum.

12. Equipment racks equipped with overtemperature alarms which remove console power within 30 seconds when any rack is in overtemperature condition. Alarm system is equipped with manual override and flashing light indicators located on status and control panels on front of low-boy consoles.

13. Confort and equipment air conditioning monitor panels provided in each room for visual and audio indication of conditioning air parameters.

d. Lighting

1. All rooms illuminated at minimum 90-foot-candle level at 30 inches above floor.

2. Recessed lamp fixtures equipped with cold cathode fluorescent lamps with leaded glass shields for RFI suppression.

3. Selected fixtures arranged for automatic transfer to standby generator.

e. Floor loading

1. 36-inch elevated floor in control, computer, and terminal facility room. Flooring made up of 18-inch by 18-inch aluminum panels (covered with vinyl tile) and stringers.

2. Elevated floor capable of carrying 1000-pound concentrated castor load at any point on a panel with maximum midspan deflection of 0.1 inch.

3. Floor cutouts and equipment tie-down provisions made on individual console, cabinet, or functional cabinet group basis.

f. Grounding (Figure A-2)

1. Four separate grounding systems are utilized: (1) building ground; (2) isolated power neutral ground; (3) static ground; (4) signal ground.

2. Building ground and power ground terminate at common point on facility earth ground mat but are otherwise isolated from each other.

3. Signal and static ground terminate at one point on the instrumentation earth ground but are otherwise isolated from each other.

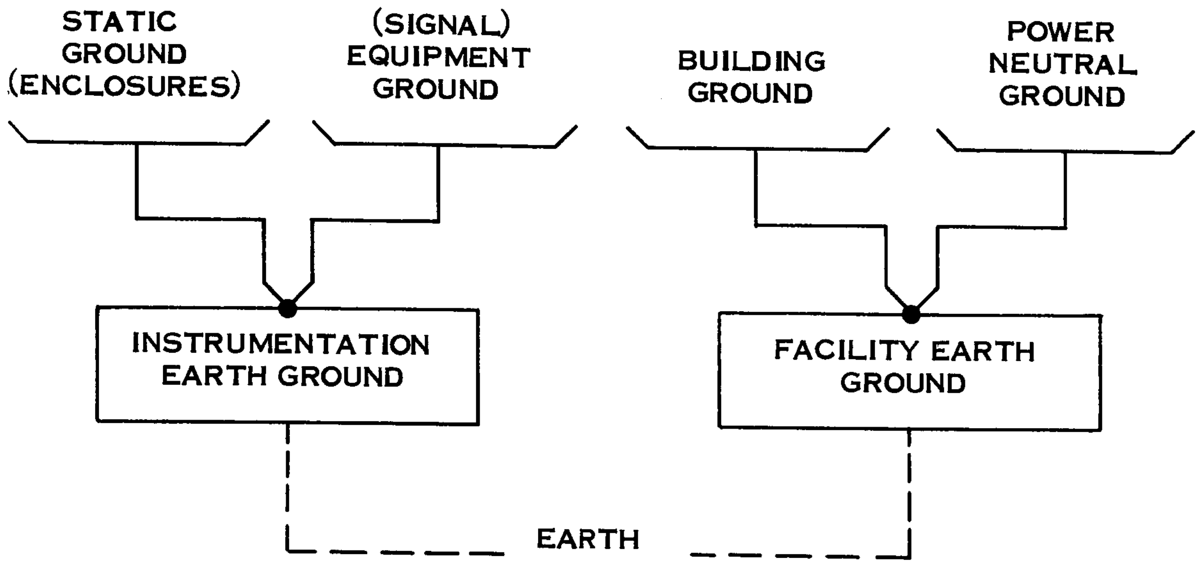


Figure A-2. Ground Terminations

APPENDIX B
SPACECRAFT INTERFACE EQUIPMENT

B.1 INTRODUCTION

The following is a brief description of the major pieces of Spacecraft Interface Equipment (SIE) required to support factory and KSC acceptance tests. The spacecraft design used to generate this list was the preferred design in GE's Task B study for JPL. A listing of the controls and monitors required to support these is also given which will give the OSE load on the control and data paths and also indicates the potential utilization and flexibility required to support the tests planned. It should be noted that the control signals are to be verified by a separate verification link and are, therefore, not a part of the monitors. A listing of the test points (umbilical identified by *) and their usage is also included.

B.2 POWER SUBSYSTEM (BAYS 1-2-16)

B.2.1 GROUND POWER SUPPLY

This supply is used to supply power to the spacecraft in place of the solar array. As such, it should duplicate the solar array characteristic V-I curve under the sun conditions ranging from 1.2 Earth to 0.8 Mars illumination. The V-I curve should be adjustable from best to worst case array simulation. The voltage should also be capable of being lowered gradually to a point below the battery level so as to cause power transfer. This unit is required for launch operations.

B.2.1.1 Controls

a. Voltage level adjustment (8 bits)
b. Internal impedance adj (4 bits)
c. Overvoltage trip point (4 bits)
d. Overcurrent trip point (4 bits)

e. Ac on/off

f. Output on/off

B.2.1.2 Monitors

a. Voltage (10 bits)

b. Current (10 bits)

c. Relay/trip status (4 discretes)

B.2.2 GROUND POWER BACKUP

This battery is required to allow an orderly shutdown of the spacecraft in the event of a power failure, either ac power or the power supply of B.2.1. The battery level should be higher than the maximum expected spacecraft battery level. It should share the over-current and output switching of the ground power supply (B.2.1 above). This unit should be switched on line as a result of a sensed low output of the ground power supply. This unit is required to support launch operations.

B.2.2.1 Controls

a. Enable/disable

b. Transfer setting (4 bits)

B.2.2.2 Monitors

a. Voltage (10 bits)

b. Transfer setting

c. Status (3 discretes)

B.2.3 ARRAY/BATTERY BUS ENABLE/DISABLE

This unit controls the spacecraft's Array/Battery Bus motor driven contacts that apply the batteries or the arrays to the main spacecraft bus. This switch does not have to be closed to allow ground power to be applied, but is required to charge batteries. The disable function should be executed when an overcurrent is sensed in B.2.1 and should be executed from a power fail safe source. This unit is required for launch operations.

B.2.3.1 Controls

a. Enable/disable
b. Voltage setting (4 bits)

B.2.3.2 Monitors

a. Voltage
b. Backup battery voltage
c. Current
d. Status (3 discretes)

B.2.4 BATTERY SIMULATORS (3 REQUIRED)

These are required to allow tests of the battery charging circuits, load sharing circuits, etc., to be conducted over the entire range of potential battery characteristics and to allow testing to proceed without the use of expensive batteries. Each should have over-voltage and over-current trip circuits. TLM temperature sensors are to be simulated with three levels of resistances.

B.2.4.1 Controls Per Simulator

a. Voltage (10 bits)
b. Internal Impedance (8 bits)

c. Charge characteristic (8 bits)

d. Ac on/off

e. Dc on/off

f. Temperature simulator (3 states)

B.2.4.2 Monitors Per Simulator

a. Voltage (10 bits)

b. Discharge current (10 bits)

c. Charge current (10 bits)

d. Status (5 discretes)

e. Temperature sampling current (10 bits)

B.2.5 SOLAR STRING SIMULATOR

This simulator is needed to simulate the output of a solar string (i.e., 1/60 of the solar array). It is to be switched into any one of the 60 zener monoblocs on the solar array side. Its characteristics must allow the verification of the zener monoblocs, and load carrying capacity of the circuits.

B.2.5.1 Controls

a. Input selection (one of 60--others grounded)

b. Voltage (10 bits)

c. Ac on/off

d. Dc on/off

B.2.5.2 Monitors

a. Voltage (10 bits)

b. Current (input) (10 bits)

c. Current (ground loop) (10 bits)

d. Selected point

e. Status (5 discretes)

B.2.6 SOLAR STRING SIMULATOR

This is to provide a light source capable of illuminating a solar string (1/60 of the array) and a variable load (to be connected to the zener monobloc side as in B.2.5 above) such that the V-I curve can be obtained for various levels of illumination. The test will be of short duration; cooling will not be required. OSE temperature sensors will be supplied.

B.2.6.1 Controls

a. Intensity (8 bits)

b. Load (4 bits)

c. Lamp on/off

B.2.6.2 Monitors

a. Voltage (10 bits)

b. Current (10 bits)

c. Temperatures (2) (8 bits)

d. Load

e. Status (2 discretes)

B.2.7 CLOCK

Provides a substitute clock that is plugged into the spacecraft timing countdown chain at the 268.8 KHz point. The frequency is to be varied to accomplish tests associated with the 2.4 KHz supply, TLM timing, Data Storage Tape Unit drives, etc. Range should be wide enough to cause switch over to a backup oscillator.

B.2.7.1 Controls

a. Frequency (10 bits)
b. Enable/disable
c. $\Delta f/t$
d. Frequency limits

NOTE (c-d may be computer controlled by controlling (d) as a function of time)

B.2.7.2 Monitors

a. Level (8 bits)
b. Frequency (20 bits)
c. Status (2 discretes)

B.2.8 DC LOAD BANK

Resistive load banks are to be supplied that can be switched to the various dc busses in the spacecraft (power subsystem buses and outputs of the other subsystem T/Rs). The load should be such as to load the supplies/regulators down such that they see 1.2 times maximum expected load. Several banks will be required in order to limit load on any supply.

B.2.8.1 Controls

a. Load (8 bits)
b. Output select (one of 50)
c. On/off

B.2.8.2 Monitors

a. Voltage (10 bits)

b. Current (10 bits)

c. Status (5 discretes)

B.2.9 400 Hz LOAD BANK

This would be essentially as B.2.8 above except to be connected to one of the 7-400 Hz lines. Inductive loads will be used in addition to the resistive loads.

B.2.10 LOAD BANK (2.4 KHz)

This would be essentially as B.2.9 above except it would be connected to one of the two 2.4 KHz lines.

B.2.11 SIGNAL SOURCE

A signal source will be required to introduce biases into the power subsystem fault sensors at enough points to cause operation of all modes of fault detection, and also for applying a bias to the main regulator magnetic amplifier in order to cause the regulated voltage to be raised or lowered (note: this will cause the 2.4 KHz and 400 Hz output voltages to vary also) so that the entire spacecraft can be operated under varying voltage conditions.

> NOTE: This may require more than one supply depending on the potential spacecraft damage which could be caused by supplying an improper signal level to one of the spacecraft points.

B.2.11.1 Controls

a. Level (10 bits)

b. Test point select (one of 14)

c. On/off

B.2.11.2 Monitors

a. Voltage (10 bits)
b. Current
c. Selection point
d. Status

B.2.12 TRANSIENT DETECTORS

Transient detectors (High and Low) will be connected continuously to all of the spacecraft buses (includes all outputs of all other subsystem T/Rs). Upon detecting a transient, a steady-state signal will be generated until reset. A few will require variable limits.

B.2.12.1 Controls

a. High limits (5--4 bits each)
b. Low limits (5--4 bits each)
c. Reset

B.2.12.2 Monitors

a. Status (100 discretes)

B.2.13 SQUIB SIMULATORS

The Pyro subsystem will require approximately 60 squib simulators that are to be inserted into the spacecraft at the location of the normal squib. The squib simulators will be monitored and reset via the continuity loop.

B.2.13.1 Controls

a. Trip point (manually set)

b. Reset

B.2.13.2 Monitors

a. Resistance

B.2.14 SPACECRAFT CONNECTIONS

The following is a listing of the spacecraft connections and their use required for system tests of the Power and Pyro subsystems (Bays 1-2-16).

*Indicates umbilical connection--numbers in parenthesis indicate number of points less returns. Where the spacecraft TLM contains the same data point, the direct access usage will also include calibration of the signal conditioners by comparing OSE measured value vs TLM value.

	Connection	Usage
a.	Ground Power (1)*	Ground power supply (B.2.1) to be connected to this point. Battery charging to be accomplished via this connection. Control and measure voltages and spacecraft currents for test purposes.
b.	Array/Battery Bus Enable (2)*	Controls spacecraft power on/off from batteries or solar string. Not required for ground power. Required for battery charging.
c.	Solar Cell Strings (60)	Use in conjunction with (B.2.5) and (B.2.6) above.
d.	Battery Power Connector (3)	This is not a test connection but rather the battery simulators (B.2.4) are plugged into this spacecraft point.

e.	Battery TLM Temp (3)	As in (d) above.
f.	Clock (3)	Used in conjunction with B. 2. 7 above.
g.	Raw Bus A-B (2)	Used to measure raw bus A and B voltage - detect high or low transients in conjunction with transient detector (B. 2. 12) above.
h.	Regulated Bus (2)	As in (g) above except for regulated buses. These lines are also to be used for loading the regulators by the dc Load Bank (B. 2. 8 above).
i.	2. 4 KHz Bus (2)	As in (h) above except load bank is (B. 2. 10) above. Measure frequency, wave shape analysis, etc.
j.	400 Hz Bus (7)	As in (i) above except load bank is (B. 2. 9) above.
k.	Fault Sensor Stimuli (12)	Test point for inserting bias supply (B. 2. 11) above to cause "faults."
l.	Regulator Bias (2)	Test point for inserting bias signal (B. 2. 11) above to cause various voltage levels throughout the spacecraft via 2. 4 KHz supply and the various unregulated subsystem T/R supplies.
m.	Command Detector (1)	Monitor (by detecting pulse) that a command was received from the Command subsystem or from the C&S subsystem. Measure pulse characteristic. (Monitor is in common path of relay circuits operated by commands.)
n.	Pyro T/R Outputs (8)	As in (g) above.
o.	Pyro Capacitor Voltage (8)	Monitor capacitor voltage to establish charge and discharge time constants.
p.	Pyro Continuity Loop (1)*	Used in conjunction with B. 2. 13 when squib simulators are employed. To be used to monitor "safety" and "plugs in" condition when squibs are installed.

B. 3 DATA STORAGE SUBSYSTEM (BAYS 7-8)

B. 3. 1 MTR TEST UNIT

This unit will be plugged into one Magnetic Tape Recorder (MTR) at a time. It will modify the record signal of each of the 14 data record heads in an identical manner and the timing record head in a like manner, but independently. This unit will be inserted directly into the MTR (or its electronics) under test. This unit will be used to measure the record/playback margin. It will also be used to measure the normal signal characteristics.

B. 3. 1. 1 Controls

a. Data loop (8 bits)
b. Timing loop (8 bits)

B. 3. 1. 2 Monitors

a. Record signal (15)

B. 3. 2 SPACECRAFT CONNECTIONS

a.	T/R Outputs (9)	Used to measure T/R voltages and in conjunction with the Transient Detector (B. 2. 12) detect any transients. Load bank (B. 2. 8) will be used to load the T/R.
b.	Phase Lock Detector (6)	Measure speed compensation error as Clock (B. 2. 7) is varied during record and playback.
c.	Command Detector (1)	Monitor (by detecting a pulse) that a command was received from either the Command or C&S subsystem.
d.	MTR Test (15)	Points for connecting (B. 3. 1) above to the circuits.

B.4 TELEMETRY SUBSYSTEM (BAY 9)

B.4.1 CALIBRATION SOURCE NO. 1

An adjustable source (0-3.2 volts) with at least 10 bit resolution. Source impedance shall be varied to cover 1.5 times the range specified for A/D converter. Unit will be connected to the OSE 3.2-volt data point of the TLM format, and will be used to establish the bit weighting characteristics of the spacecraft TLM A/D converters.

B.4.1.1 Controls

a. Voltage (10 bits)
b. Impedance (4 bits)
c. On/off

B.4.1.2 Monitors

a. Voltage (10 bits)
b. Status (2 discretes)

B.4.2 CALIBRATION SOURCE NO. 2

As B.4.1 above, to cover the range of -1.6 to +1.6 volts. Will be connected continuously to each OSE point required to calibrate each 1.6 bucking TLM supply.

B.4.3 CALIBRATION SOURCE NO. 3

Variable resistances (3 values minimum) that cover the range of 500 to 600 ohms. To be used to measure the constant current supplies and measure the 0.5 volt bucking supply and 32 x amplifier.

B.4.3.1 Controls

a. Resistance (3 bits)

B. 4. 3. 2 Monitors

a. Voltage (10 bits)

b. Current (10 bits)

B. 4. 4 WORD SYNCHRONIZER

Operating on real time digital data streams A and B, obtain bit synch and word synch for each independently. Word synch will be established by detecting frame synch (14 consecutive 1's followed by a 0) and then dividing bit rate by 7, counting the initial 0. Data will be outputted to the OSE PCM TLM data stream to be returned to computer group for decommutation. The pulse characteristics are to be obtained. This unit is required for launch operations.

B. 4. 4. 1 Monitors

a. Digital data (two 10-bit words)

b. Wave shape analysis (amplitude, rise/fall time, phase A/B)

B. 4. 5 SPACECRAFT CONNECTIONS

a.	Data Streams A&B* (2)	The two digital data streams will be decommutated. Pulse shape measurements, analysis, etc. will be accomplished using B. 4. 4 above.
b.	T/R Outputs (4)	Measure voltage and, using (B. 2. 12), detect transients and (B. 2. 8) provide additional loading.
c.	3. 6 v OSE Data Point (1)	TLM data point used to allow calibration of each A/D converter by using B. 4. 2 above.
d.	±1. 6v OSE Data Point ( )	TLM data point to allow calibration of 1. 6 bucking supply by using B. 4. 2 above. One point required for each supply.

e.	OSE Resistance Points	TLM data points used to allow calibration of each 0.5 volt bucking supply and 32 x amplifier by using B.4.3 above.
f.	Command Detector (1)	Monitor (by detecting a pulse) that a command was received by the TLM subsystem from either the Command or C&S subsystem.

B.5 COMMAND SUBSYSTEM (BAY 10)

B.5.1 COMMAND MODULATOR

The command modulator is to output to the spacecraft via the Radio subsystem OSE or directly to the spacecraft Command Subsystem via an umbilical point, a signal of the form $PN \oplus 2f_s \pm \cos 2\pi 2f_s t$ where $+ \cos 2\pi 2f_s t$ represents a data sub bit 1 and $-\cos 2\pi 2f_s t$ represents a data sub bit 0. The data will be obtained from a remote source. The modulator will receive via an umbilical point, the spacecraft Command Subsystem detected sub bits (available at end of sub bit transmission time). If the detected sub bit is improper, the modulator will switch to an inhibit mode which will output a series of identical sub bits. There will be two $2f_s$ rates which will be selected and varied as desired. The following control listing gives an idea to the extent of the variations required in the data format in order to test the Command Subsystem circuits. This unit is required to support launch operations.

B.5.1.1 Controls

a. Data (64 bits)
b. Preamble data (6 bits)
c. Decoder select (2 bits)
d. Start/stop PN
e. Start/stop data
f. $2f_s$ rate
g. $2f_s$ (8 bits)

h. PN$\oplus$2f_s power (4 bits)

i. cos 2f_st power (4 bits)

j. PN$\oplus$2f_s rise/fall time degrade (4 bits)

k. cos 2f_st wave shape degrade (4 bits)

l. No. of consecutive PN errors (5 bits)

m. Starting point of (ℓ) (5 bits)

n. Repeat (ℓ) n times (4 bits)

o. Phase shift PN and data n 8f_s bits (8 bits)

B. 5. 1. 2 Monitors

a. Spacecraft error detected

b. Data sent (64 Bits)

c. Manual and local analysis of wave shape

B. 5. 2 SPACECRAFT CONNECTIONS

a. T/R Outputs (3)	Measure voltage and use in conjunction with B. 2. 12 to monitor high and low transients. Load T/R output by use of B. 2. 8.
b. Ground Command Enable* (1)	Enables data path for spacecraft commands via (c) below.
c. Ground Command* (1)	Carries the spacecraft command data via the umbilical from (B. 5. 1) above.
d. Detected Sub Bits* (3)	Used by B. 5. 1 to verify reception spacecraft commands received either via the umbilical or via RF.
e. PN Synch (3)	Used to detect when each of the three spacecraft detectors establish or lose PN synch.

f.	Decoder A Inhibit (1)	Inhibits decoder A matrix interrogate pulse thus allowing testing of modes of decoder bypass and verifying proper decoder operation in non-bypass modes.
g.	Decoder B Inhibit (1)	As in (f) above.
h.	Command Executed (1)	Indicates that a command was executed by the command subsystem. This is not a logic signal, but the drive signal common to all primaries of the isolation transformers.

B.6 RADIO SUBSYSTEM (BAY 11-12)

B.6.1 TRANSMITTER

Provides a remotely variable frequency and power RF transmitter. A ranging signal with variable parameters will be able to be introduced. Command data from the Command Modulator (B.5.1 above) will also be able to be introduced.

B.6.1.1 Controls

a. Frequency (10 bits)
b. Power (8 bits)
c. Ranging data parameters (4-10 bits)
d. Enable command (1)
e. On/off (1)

B.6.2 RECEIVER

Provides means for receiving the spacecraft transmitted signals, demodulating the ranging signal and the TLM data signal. The TLM engineering data signal should be in a form compatible for input into the TLM OSE Word Synchronizer (B.4.4 above). The TLM scientific data will be digitized and routed to the Science OSE directly. This unit will be manually controlled.

B. 6. 3 RF TEST EQUIPMENT

Provides complete capability to analyze the transmitted and received signals to establish power, frequency modulation index, spectrum analysis, wave shape analysis, etc. Power and frequency meters should be connected continuously to the transmitter and receiver. The others can be time shared.

B. 6. 3. 1 Controls

a. All manually controlled

B. 6. 3. 2 Monitors

a. Frequency (2-20 bits)
b. Power (2-8 bits)
c. Remainder (manually monitored)

B. 6. 4 RF LINK

Provides means for connecting and switching the transmitter and receiver to OSE antennas for RF transmission to the spacecraft or via wave guide/coax lines to the spacecraft antenna connectors.

B. 6. 5 SPACECRAFT CONNECTIONS

a. T/R Outputs (6)	Measure voltage and use in conjunction with B. 2. 12 to detect high and low transients. Load T/Rs output by use of (B. 2. 8).
b. Receivers AGC (3)	Monitor as RF signal, power, etc is varied.
c. Receiver SPE and DPE (6)	Monitor as RF signal, power, etc is varied.
d. Power amp - exciter levels (6)	Monitor as power, etc is varied.

e.	Antenna (3)	Provides means for connecting B. 6. 1 and B. 6. 2 directly into the spacecraft via B. 6. 4.
f.	Command Detector (1)	Detects that a command was received from either the Command or C&S Subsystems.
g.	Composite Command Signal (3)	Provides means for analyzing and comparing (manually) the OSE transmitted command signal and the spacecraft detected composite command signal.
h.	Combined TLM signal (3)	Provides means for analyzing and comparing (manually) the combined TLM signals into the Radio Subsystem and those received on the ground.

B. 7 COMPUTER AND SEQUENCER SUBSYSTEM (BAY 14)

B. 7. 1 C&S CLOCK

Provides a means for inserting a high speed clock into the C&S subsystem. The speeds will be 10-100-1000-10,000 x normal as selected. The high speed timing will be stopped by a settable pulse counter sensing C&S clock pulses.

B. 7. 1. 1 Controls

a. Speed (4 ranges)
b. Stop time (20 bits)
c. Reset "stop" counter
d. Start/stop

B. 7. 1. 2 Monitors

a. Elapsed time (20 bits)

B.7.2 SOLAR ASPECT SENSOR STIMULATORS

Provides a means for illuminating the solar aspect sensors mounted on Bay 14. The light source will be manually adjusted about two axes. Position readoffs should be available.

B.7.2.1 Controls

a. Illumination on/off

B.7.2.2 Monitors

a. Position 2 axis (8 bits each)

B.7.3 PN/DEC CONVERTER

Performs conversion of decimal to PN code or PN code to decimal. The PN code generator will be selected from one of four possible combinations (length, feedback, etc).

B.7.3.1 Controls

a. PN Code Set (25 bits)
b. Dec Code Set (25 bits)
c. Format select (one of four)

B.7.3.2 Monitors

a. PN Readout (25 bits)
b. Dec Readout (25 bits)

B.7.4 SPACECRAFT CONNECTIONS

a.	T/R Outputs (4)	Measure voltage and use in conjunction with (B.2.12) to detect high or low transients. Load T/Rs output by use of B.2.8.
b.	OSE Clock (1)	Input OSE C&S clock (B.7.1) above.
c.	Enable OSE Clock (1)	Switches C&S Subsystem clock source from spacecraft source to OSE source (B.7.1).
d.	C&S Clock (1)	Used to accumulate C&S time regardless of source used by (B.7.1) to establish stop time.
e.	TTG Registers (8)	Allows sampling one register of each of the eight different Time To Go Registers. This is required to speed up testing. (TLM sampling rate is prohibitively low).
f.	Memory Read Out (1)	Similar to (e) above, except reading out memory.
g.	Separation Switch Bypass* (1)	Bypasses separation switch and allows TTG registers to be advanced by the C&S clock and allows their setting to be verified.
h.	Command Execute (1)	A pulse that indicates a command was put out from the C&S subsystem decoder as B.5.2(h) above.
i.	High Gain Antenna Drive Motors (2)	Detect, measure (wave shape analyses), and count etc. the pulses to the HGA drive motor. Obtain time delays, pulse widths, amplitudes, counts, etc. Apply a transient detector during normally quiet periods.

B.8 <u>GUIDANCE AND CONTROL (BAY 15)</u>

B.8.1 STAR SENSORS STIMULATORS (2 REQUIRED)

Provides means for stimulating each of the star sensors with a variable position (both cone and clock axis), variable rate, variable intensity and size star simulator. The sun shade sensing circuits will be stimulated by variable intensity light.

B. 8. 1. 1 Controls/Stimulator

a. Rate (4 bits each axis)
b. Intensity (10 bits)
c. On/off
d. Start/stop (each axis)
e. Sun Intensity (10 bits)

B. 8. 1. 2 Monitors/Stimulator

a. Position (each axis) - (10 bits)
b. Intensity (10 bits)
c. Limit Discretes (4 bits)
d. Sun Intensity (10 bits)

B. 8. 2 SUN SENSOR STIMULATORS (24 REQUIRED)

Provides means for stimulating each group of sun sensors (acquisition, cruise, sun gates, etc.) with variable intensity light. Apparent position and motion will be inserted into the system by varying the intensity on the various sensors.

B. 8. 2. 1 Controls/Stimuli

a. Intensity (8 bits)
b. On/off

B. 8. 2. 2 Monitors/Stimuli

a. Intensity (8 bits)
b. Temperatures (4 bits)

B. 8. 3 GYRO TORQUERS (3 REQUIRED)

Provides means for supplying variable current to each gyro torque motor to cause apparent spacecraft rates. There shall be provisions to sense the Gyro Amplifier output and limit the gyro signal if a oversize value is sensed.

B. 8. 3. 1 Controls/Torquer

a. "Rate" (10 bits)
b. On/off
c. Limit (8 bits)
d. Gyro Package Select

B. 8. 3. 2 Monitors/Torquer

a. Current (10 bits)
b. Integrated current (10 bits)
c. Limiting status
d. Gyro amp output (10 bits)
e. Integrated acceleration (10 bits)

B. 8. 4 NOZZLE FLOW DETECTORS

Provides means for sensing and identifying which nozzle from which gas is flowing.

B. 8. 4. 1 Monitors

a. On/off (24 discretes)

B. 8. 5 JET VANE LOAD/POSITION (16 REQUIRED)

Provides means for loading and sensing the positions of the propulsion system jet vanes.

B. 8. 5. 1 Controls/Device

a. Load Force (manual)

B. 8. 5. 2 Monitors/Device

a. Position (10 bits)

NOTE: B. 8. 1 through B. 8. 5 are to be used to perform open loop testing or simplified close loop testing. This can assume that the nozzles cause constant acceleration and that the acceleration caused by jet position is a direct function of that position. These accelerations are to be integrated to establish the rate which in turn is integrated to establish position. (Item B. 8. 3 provides this integrate function.) The position information is to be fed into the star and sun sensors and the rate into the gyro.

B. 8. 6 COLD GAS SUPPLY

Provides means for supplying variable pressure N_2 gas such that the gas system can be tested under various upstream pressures and flow conditions. Note: Proof pressure and leak tests are to be carried out in a special area and that equipment is not part of STE.

B. 8. 6. 1 Controls

a. Pressure (8 bits)
b. Valving (8 discretes)

B. 8. 6. 2 Monitors

a. Pressure (8 bits)
b. Flow (8 bits)

B. 8. 7 SPACECRAFT CONNECTIONS

a.	T/R Outputs (5)	Measure voltages and use in conjunction with B. 2. 12 to detect high and low transients. Load T/R output by use of B. 2. 8.
b.	Gyro Torque Signals* (6)	Path for inserting gyro torquer signal from B. 8. 3 above.
c.	Gyro Amplifier Outputs* (3)	Signal to calibrate gain of amplifier and gyros. Used to prevent excessive gyro error signals. Used to verify operation of summing circuits at input to threshold detector and autopilot.
d.	Acquisition Sun Sensor Outputs (12)	Monitors position error signal caused by B. 8. 2 above. Used to calculate transfer function of sensors and using (f) below, the gain of acquisition amplifiers and rate limiting circuits.
e.	Cruise Sun Sensors (6)	As in (d) above, plus verification of commanded gain changes.
f.	Threshold Detectors (9)	Monitor inputs to three threshold detectors per channel. It is used in conjunction with c, d, e above to establish gains and derived feedback signal characteristics. It will be used with (g) to establish trip point of each detector.
g.	Solenoid Valve Commands (24)	Monitor solenoid valve commands and obtain solenoid valve signatures. Used in conjunction with (f) to establish threshold trip point.
h.	Cruise Sun Sensor Amp Gain Control (6)	Provides an OSE override of command control of the gain of each of the three pitch and yaw amplifiers. By varying the gains, the operations of the threshold amplifiers and their majority logic can be verified independently.
i.	Autopilot Outputs (10)	Provides means for verifying open loop gain of the autopilot (using c above as input) and thrust vector control (using B. 8. 5 above).

j. Regulated Pressures (2) — Monitor spacecraft-regulated pressure transducers to obtain steady state and transient operation under various flow and upstream pressures.

k. Command Detector (1) — Detect pulse indicating that a command was received from either the command subsystem or the C&S subsystem.

l. Logic Control Unit (8) — Provides logic signals indicating mode of G&C subsystem.

B.9 STRUCTURE

B.9.1 HIGH GAIN ANTENNA DRIVE

Provides means for simulating the antenna inertia load about two axes such that the deployment and drive capability can be tested. Measures the position and drive accelerations about each axis.

B.9.1.1 Monitors

a. Position (2 axes) 10 bits

b. Acceleration (2 axes) 8 bits

c. Deployed/stowed (2 discretes)

B.9.1.2 Controls

a. Load (manually adjust)

b. Deploy (pull manually pin puller)

B.9.2 THERMAL CONTROL

Provides means for evaluating the operation of the thermal louvres by attaching OSE position and temperature sensors and loading devices to them and obtaining position vs temperature under various loading conditions.

B.9.2.1 Controls--Manual

a. Louvre load
b. Temperature control via heat lamp

B.9.2.2 Monitors

a. Position (10 bits)
b. Temperatures (10 bits)

B.9.3 SPACECRAFT COOLING

Provides means for cooling of the spacecraft as required to maintain the desired internal temperatures. This unit will be a self-contained and closed-loop control system using its own temperature sensors and flow controls, etc.

B.9.3.1 Controls

a. Set desired temperature (8 bits)

B.9.3.2 Monitors

a. Position (10 bits)
b. Temperatures (16)
c. Alarm

B.9.4 SEPARATION SWITCH ACTUATOR

Provides means for operating the separation switches from a remote location and for measuring the trip point and hysteresis of the switch locally.

B.9.4.1 Controls

a. On/off

B.9.5 SEPARATION MECHANISM

Provides means for testing the adapter structure, including the gas supply and the plunger mechanisms. There will be switches to detect plunger operation and to verify timing of simultaneous operation.

B.9.5.1 Controls

a. Gas supply controls (5 discretes)
b. Actuation

B.9.5.2 Monitors

a. Flow (8 bits)
b. Pressures (8 bits)
c. Actuation (16 discretes)

B.9.6 SPACECRAFT CONNECTIONS

a. Heaters (20)	Monitor operation of the spacecraft thermostats and obtain operating and reset temperatures. They will also be used to bypass the normal thermostats in order to raise the temperature to cause the backup thermostats to operate.

B.10 PROPULSION

B.10.1 GAS SUPPLY

The propulsion subsystem will be tested by flowing gas through the various paths and observing proper operation under those conditions. The operation of the thrust vector control is covered under the G&C equipment.

B.10.1.1 Controls

a. Pressure (2-8 bits)

b. Valving (15 discretes)

B.10.1.2 Monitors

a. Pressures (5-8 bits)

b. Flow (8 bits)

B.10.2 SPACECRAFT CONNECTIONS

a.	Tank Temperatures* (4)	Monitor tank temperatures for safety purposes in Launch Mode.
b.	Tank Pressures* (6)	As in a above.
c.	Tank Vent/Dump* (6)	Provides means for venting or dumping propellants when in the launch mode.
d.	Pressures (8)	Provides means for measuring pressures of various points and for detecting pressure transients etc.

B.11 SCIENCE (INCLUDING PSP AND DAE) - ALL GFE

B.11.1 SCIENCE/DAE

Stimulation of the science equipment will be provided so as to calibrate their operation. This will include such things as several TV test patterns with various levels of illumination and filters, controlled UV sources, etc. Special purpose logic units will be required to verify data compression and error-correcting encoder elements of the DAE.

B.11.1.1 Controls

a. Levels 8 (8 bits)

b. Discretes (15)

B.11.1.2 Monitors

a. Analogs 10 (8 bits)

b. Discretes (20)

B.11.2 PSP CONTROL LOOP

Stimulation for the planet sensor and mechanisms for loading the drive mechanisms and monitoring their positions will be required.

B.11.2.1 Controls

a. Levels 5 (8 bits)

b. Discretes (10)

B.11.2.2 Monitors

a. Analog 5 (8 bits)

b. Discretes 5

B.11.3 SPACECRAFT CONNECTIONS

a. Estimated 25 functions	These would be required to monitor the science package's T/Rs for normal voltages, detect transients, etc. Means would also be required to obtain significant data points in each instrument package such that its operation can be assessed. Typical of this would be TV analog signals in real time - data compressor input and output.

B.12 CAPSULE SIMULATOR

This unit will provide the means for verifying the spacecraft-to-capsule interface including provisions for stimulating the relay radio link. The interface will be verified under nominal to worst case conditions. The following is a summary of the interfaces to be simulated.

a. Relay radio transmitter
b. Power loads
c. TLM synch signals detection
d. TLM data out (fixed format)
e. Command signal detection
f. Umbilical circuits

B.12.1 CONTROLS

a. RF power
b. RF frequency
c. Data characteristics (3)
d. Load bank
e. TLM data characteristics (3)
f. TLM synch circuits loading
g. Command circuits loading

B.12.2 MONITORS

a. Voltage
b. Current
c. TLM synch wave shapes
d. Command data
e. Command data wave shapes

B.13 VERIFICATION UNIT

Provides means for verifying the equipment up to the connectors that interface with the spacecraft. This unit need not simulate the operation of the spacecraft in any manner. The numbers and characteristics of the additional controls and monitors required to support this unit are estimated as follows.

B.13.1 CONTROLS

a. Analogs (20)
b. Discretes (25)

B.13.2 MONITORS

a. Analogs (20)
b. Discretes (25)

APPENDIX C

LAUNCH EQUIPMENT REQUIREMENTS

C.1 TEST AND OPERATIONS PLAN PHILOSOPHY

The General Electric Task B Test and Operations Plan (TOP) for launch pad operations (all operations following pad mating through lift-off) is based on the philosophy that -- barring failure to operate -- the spacecraft are committed to flight, the pad time is minimized, and testing for test sake is finished at the hangar. The tests will be simple go-no go; test points, etc., will be made available only for (a) those functions required to prepare for and carry out a launch, (b) provide safety monitors and controls, and (c) provide test capabilities when a significant system gain is provided. The spacecraft will be delivered to the pad area in a buttoned-up state with all batteries, squibs, igniters, etc., installed.

> NOTE: Task B study also had the spacecraft fueled and pressurized. Current thinking is that this may not be desirable -- primarily from a safety point of view.

C.2 PAD OPERATIONS

The functions required by Voyager are as follows:

a. Preparation for launch

1. Enable battery/array bus.
2. Load and verify C&S memory with command functions and time (512 words at 28 bits).
3. Load and verify C&S time to go register with PNG time codes equal to time from separation to event plus pad verification time (8 words at 28 bits).
4. Charge batteries.
5. Load and monitor fuel, oxidizer cold gas, etc.
6. Monitor safety monitors, i.e., continuity loops, temperatures, pressures.

b. Launch Operations (Final Countdown)

1. Initialize system.

   (a) Select integrating gyro package.

   (b) Select RF paths.

   (c) Select telemetry mode.

   (d) Select battery charge levels.

2. Gyro and tape recorders motors on -- required to withstand launch environment.

3. Obtain RF lock with DSS No. 71.

4. Transfer to internal power.

5. Monitor ambient status of all spacecraft subsystems.

C.3 UMBILICAL FUNCTIONS

Since all the above control or loading functions are normal functions of the spacecraft Command Subsystem and most of the monitors required are normal telemetry monitors, the spacecraft Telecommunications Subsystem in RF contact with DSS No. 71 and a remotely controlled ground power supply can perform nearly all the required functions. However, since some portion of the operations will be accomplished during RF silence, hardware (direct access) points will be required to bypass the radio link; and since the S/C bus will not be on continuously (TLM not on), separate safety monitors will be required. In addition, some additional capability will be desired. This is shown below in a minimum umbilical list.

a. Battery/Array Enable (On-Off). Controls spacecraft motor-driven contacts switching batteries and array output on-or-off the spacecraft main bus. Must be closed to charge batteries (via ground power). Need not be closed to operate spacecraft on ground power. Emergency shutdown requires this switch to be opened and ground power removed. Operation is infrequent -- high-speed response required only in case of emergency with switch closed.

b. Ground Power. Supplies power to spacecraft to perform prelaunch operations and to charge batteries. Transfer to internal power is accomplished by lowering this supply voltage until batteries assume load at which point this supply can be disconnected. Telemetry, C&S, and Command Subsystems are on with any power to spacecraft. This supply will be backed up by a battery that is switched in automatically on detecting a ground power supply output lower than requested. Ground controls and monitors will be:

1. Ac power on.

2. Output power on.

3. Dc adjust.

4. Dc voltage monitor.

5. Dc current monitor.

6. Discrete monitors.

7. Backup supply.

There will be over-under voltage trip and overcurrent protection -- all accomplished locally and automatically. Overcurrent should cause the battery/array contact to be disabled.

c. Command Detector B (Input). This is hardwire input into detector B. The signal is a combination of PN code and data bits. Command words are either 11 bits or 64 bits long. The C&S memory (512-word, 64-bit command) and time-to-be registers (8-word, 64-bit command) will be loaded via this connection. Spacecraft control and initializing will be accomplished via this path using 11 bit commands.

This same data (PN and data) can also be transmitted to the spacecraft via RF to the Radio Subsystem at 15 bps or 1/2 bps.

d. Detected Command Data. This is the serial sub-bit data as detected by the Command Subsystem, received either via hardwire (c. above) or from the Radio Subsystem. This data will be compared to the sent data on a sub-bit by sub-bit basis. If errors are found, the ground equipment will force the remaining sub-bits into a sequence such that the spacecraft check circuits will cause an inhibit and reject to be generated and prevent the execution or storage of a bad command. This will be done by making the next sub-bit sequence incorrect and then forcing the PN code bad.

e. Enable Command Detector. Switches the command detector B input to receive hardwire input from (c) above. Command is a pulse. Input removed by spacecraft command.

f. TLM Channel A. Provides hardwire serial telemetry data (500 words, 7 bits -- at multiple rates of 17, 117, and 1870 bps). Data to be recorded-decommutated-analyzed by comparing to limits based on input conditions -- distributed for display as required. Data is same as available via Radio Subsystem.

g. Telemetry Channel B. Provides hardwire serial data in real time of telemetry data going to Data Storage Subsystem. Data to be handled as in (f) above.

NOTE: Telemetry channel A and B usage can be reversed by Channel Activity Toggle command.

h. C&S Clock Enable. Enables the C&S clock to be operated on the pad (normally enabled at spacecraft-launch vehicle separation). Required to allow memory and time-to-go register readout via telemetry. Requires continuous level for operation.

i. Pyro Continuity. Continuity loop of all squib connectors, separation switches back contacts, arm safe status, etc. Used as safety indication of Pyro Subsystem. Monitored continuously.

j. Propellant Tank Temperatures (4). Provides power, signal and returns to monitor four temperatures independent of the spacecraft's Telemetry Subsystem. Monitors to be available continuously when tanks are loaded. Alarms to be given when temperature or temperature rates exceed specified limits.

k. Propellant Tank Pressures (6). As in (j) above except for pressures.

NOTE: Power and returns can be common.

l. Tanks Vent/Dump (6). Provides control for venting and/or dumping the various propellant tanks independently. Power for this function as well as monitors must be from a "fail-proof" source.

The addition of the following functions to the above minimum umbilical list would constitute the maximum umbilical list using the current TOP philosophy:

a. Gyro Torque Signals (6). Gyro torque motor signals to allow simulating vehicle rates in order to allow testing of the interfaces between G&C and the Command, C&S, and Propulsion (Thrust Vector Control) Subsystems. The control should enable ramps, step inputs, etc., to be generated.

b. Gyro Amplifier Outputs (3). Must be used in conjunction with (a) above to prevent excessive "error" signals and for verification purposes.

c. Telemetry Speed Up. Discrete command that causes the TLM bit rate to be speeded by a factor of 8 to 16.

At normal telemetry rates, it will take at a minimum of 1 hour and 40 minutes to load/verify the C&S time-to-go registers using the normal telemetry readout. This time is all telemetry readout time. This time is marginally long -- especially if faults are found that require the registers to be reloaded. In order to reduce the readout time, either special OSE circuits can be added to the spacecraft's TTG registers to allow readout or a higher speed telemetry rate can be used. Since the telemetry is already multirate, its components are not operating near their peak frequency, and other "test" functions will be improved by having a higher telemetry data rate. This approach is preferred by OSE. The high data rate should be in the order of 8 to 16 times the maximum existing rate of 116 bps or 930 to 1860 bps.

C.4 TEST REQUIREMENTS/CAPABILITY

In establishing the minimum umbilical requirements above, it was assumed that some testing would be required on the pad. If there were to be no testing, then the only umbilical requirement (other than for safety) would be the Battery/Array Enable function. In that case, battery charging, loading the C&S memory and time-to-go registers, and system initializing (setting latching relays) would be accomplished in the hangar. Any last-minute updating required would be done via the RF command link on battery power following the establishing of the RF lock shortly before lift-off. This no-testing approach is unrealistic, as it is considered important to maintain some level of capability to observe and judge spacecraft operation up to launch. This is proper, provided the system is not penalized in order to provide the capability for the demonstration to take place.

The umbilical list generated to date is felt to be a reasonable compromise. The outstanding things that cannot be tested are:

a. Pyro circuits -- squibs installed. (It is unrealistic to expect to install squibs on the pad following tests.)

b. Antenna deployment and, therefore, drive.

c. PSP deployment and, therefore, drive.

d. Scientific equipment beyond sensing ambient.

e. Attitude control sun and star sensors. To make this a worthwhile test, mounting lights and shades would be needed within the shroud -- preferrably several for each sensor. The gain of this static test (sensor and sensor amplifier one point verification) at the cost of the lights, controls, weight, etc., added to the shroud is not considered justifiable.

f. Certain RF paths due to shroud and position of antennas.

The following is a brief summary of the test capability that is present using the maximum umbilical list above and by using the variations available in the stimulation signal sources connected. These signal sources can have the same characteristics as those used during system tests. These tests maximize the use of the spacecraft Command and Telemetry Subsystems for control and monitor.

a. Power Subsystem

   1. Battery charging/discharging as governed by installed battery characteristics.

   2. Dc regulators' response to simulate array variation (ground power supply).

   3. Transfer of power from battery to array and reverse.

   4. Load sharing.

   5. Spacecraft power required in all modes.

   6. Response to all commands.

   7. Telemetry sensors.

b. Radio Subsystem

   1. Response to all commands.

   2. Telemetry Sensors.

   3. Capability to detect/reject/lock with various RF signal parameter variations, including power, frequency, command modulation characteristics, ranging, etc.

4. Measure spacecraft transmitted-signal characteristics.

c. Command Subsystem

1. Telemetry sensors.

2. Response to Radio Subsystem outputs with various PN and data parameter variations.

3. Execution of all commands except squib, igniter, HGA drive, etc.

4. Capability to detect invalid data.

5. Responses to invalid commands.

d. G&S Subsystem

1. Telemetry sensors.

2. Response to all commands.

3. Complete memory exercise and test.

4. Time-to-go registers cycle and outputs -- except motor start/stop capsule separation.

5. Execution of all discretes except squib operations.

6. Reject circuitry.

e. G&C Subsystem

1. Telemetry sensors.

2. Response to all commands.

3. Integrating gyro package, heaters, output circuits to solenoid (torque input controlled).

4. Maneuver characteristics.

5. Acquisition sequence.

6. Autopilot operation -- gyro to propulsion TVC positions.

7. Threshold detector operation.

f. Data Storage Subsystem

   1. Telemetry sensors.

   2. Responses to all commands.

   3. Response to "all record" mode.

   4. Playback sequence and options.

g. Telemetry Subsystem

   1. Response to all commands.

   2. Format verification.

   3. Data storage record/playback.

   4. Comparison of common OSE and spacecraft data points.

h. Structure Subsystem

   1. Thermal controls under various ambients (i.e., spacecraft operating, non-operating, and under various degrees of ground cooling).

The remaining subsystems (Propulsion, Science, and Capsule) would be monitored via telemetry for their static ambient conditions.

The above functions, etc., are on a per-spacecraft basis and must be duplicated for each spacecraft.

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1.0 PROPELLANT HANDLING EQUIPMENT

1.1 Fuel

SPS Fuel Bleed Measuring Equipment (NAA), S14-124-101

Fuel Transfer and Conditioning Unit, S14-008-201 (NAA), LSC-430-94008 (GAEC)

Fuel Ready Storage Unit, S14-058-0001 (NAA), LSC-430-94058 (GAEC)

Fuel Vapor Disposal Unit, S14-060 (NAA), LSC-430-94060 (GAEC)

Apollo S/M RCS Fuel Servicing Unit, S14-063 (NAA), LSC-430-94063 (GAEC)

Apollo C/M RCS Fuel Servicing Unit, S14-064 (NAA)

Weigh Tank Calibration Unit, LDW 430-6190-1 (GAEC)

Fuel Valve and Control Boxes (NAA)

1.2 Oxidizer

Oxidizer Transfer and Conditioning Unit, S14-002-201 (NAA)
LSC-430-94002 (GAEC)

Apollo RCS Oxidizer Servicing Unit, S14-057 (NAA), LSC-430-94057 (GAEC)

Oxidizer Vapor Disposal Unit, S14-061 (NAA), LSC-430-94061 (GAEC)

SPS Oxidizer Bleed Measuring Equipment, S14-122 (NAA)

Oxidizer Valve and Control Boxes (NAA)

APOLLO FUNCTION SUMMARY

The unit receives fuel bled from SPS engine system during servicing and provides a quantitative measurement of fuel received in the unit. This unit is compatible with ACE.

APOLLO REQUIREMENTS

A. 30-gallon fuel capacity.

B. Purge capacity with GN_2.

C. Remote and local readout capability.

VOYAGER REQUIREMENTS

Current planning for loading fuel and oxidizer into the Voyager Spacecraft does not utilize the method of total filling the tanks and then "off-loading" a measured amount. Based on those requirements, this unit is not usable in Voyager. However, if the loading concept changes, this item should be reviewed.

S14-124-101	SPS FUEL BLEED MEASURING EQUIPMENT (NAA) PART NO. G14-849860-101

EQUIPMENT SUMMARY DATA SHEET
Sheet 1 of 82

APOLLO FUNCTION SUMMARY

Transfers predetermined quantities of thermally conditioned and filtered fuel (50-50, UDMH - N_2H_4) from ground storage into the Spacecraft SPS fuel tanks. The unit is mobile and is located on the pad level interfacing with the base of the service structure propellant lines, can be remotely controlled, and has the capability of unloading fuel from the Spacecraft and purging the SPS. The unit consists of three major subsystems: a pumping and flow control subsystem, a thermal conditioning subsystem, and a filtering subsystem. Components of these subsystems include pumps, valves, flow meter, pressure and temperature sensors and indicators, micronic filters, heat exchanger, heater and chiller.

APOLLO REQUIREMENTS

A. The unit is capable of cooling 22,000 pounds of propellant from 105 degrees F. to 70 degrees F. in a period of eight hours or less when the propellant is gaining heat from an external source at a rate of 100,000 British Thermal Units (Btu) per hour.

B. Heating 22,000 pounds of propellant from 35 degrees F. to 70 degrees F. in a period of 8 hours or less when the propellant is losing heat to an external source at a rate of 50,000 Btu per hour.

C. Conditioning propellant to any temperature between 35 degrees F. to 140 degrees F. (The temperature to which the unit is conditioning the propellant will be a controlled thermostatic adjustment at the unit.) The temperature of the propellant leaving the unit during this operation is maintained within 3 degrees F. of any preset temperature between 35 degrees F. and 140 degrees F.

D. Circulates fuel at 55 to 75 gpm and 200 pounds per square inch (psi). Transfers fuel to fuel loading and control unit at 55 to 75 gpm and 450 psi.

E. Measurement accuracies are as follows:

(1) Pressure - within 2 percent of the highest pressure indicated by the system.

(2) Temperature - within 1.8 degrees F. between indicated readings at 60 degrees F. to 80 degrees F. When measuring temperature outside the range 60 degrees F. to 80 degrees F., the measuring systems are accurate to within 2 percent of the highest temperature indicated by the system.

(3) Flow total and spacecraft load indicator system accuracy: Within $\pm$ 0.2 percent for any delivered quantity over 1,000 pounds.

D-3A

FOLDOUT FRAME

VOYAGER REQUIREMENTS

A. Fuel mass - approximately 4,760 pounds.

B. Temperature range - 30 degrees F. to 90 degrees F.

C. Accuracy of temperature measurement - $\pm$ 2 1/2 degrees F.

D. Accuracy of fuel weight measurement is within 0.07 percent

E. Loading time, including time to make all connections, cali evacuating, filling and disconnection, estimated to be 16 hours.

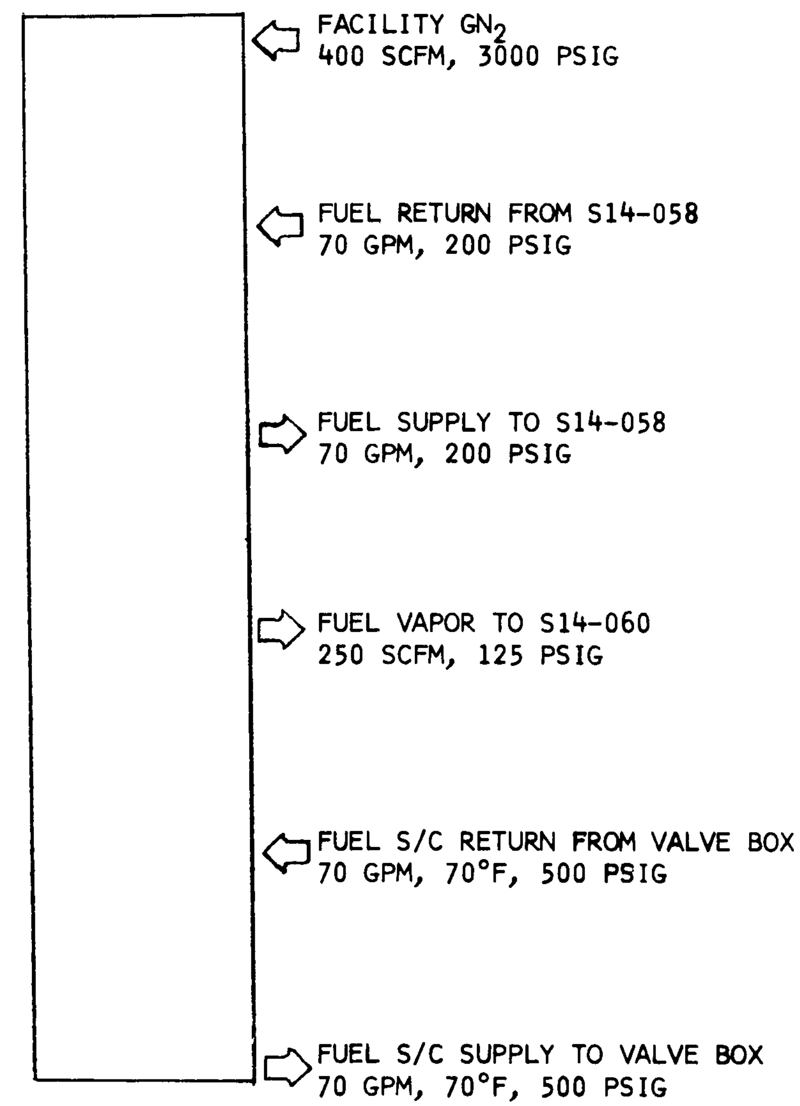
PHYSICAL SIZE

Length - 180 inches. Height - 130 inches.

Width - 90 inches. Weight - 20,000 pounds.

SUPPLY REQUIREMENTS

480 volts, 3 phase, 60 cps, 3 wire, 166 kilovolt amperes (kva) 300 psi, flow up to 400 scfm at ambient temperature.



S14-008 FUEL TRANSFER AND CONDITIONING UNIT

D-3B

FOLDOUT FRAME

of full load.
brations, purging,

. GN_2, 200 to

VOYAGER EVALUATION

The Apollo Fuel Transfer Conditioning Unit is applicable for use on Voyager. The weight of fuel required and the conditions under which it is furnished are compatible. The accuracy of fuel weight measurement for Voyager is more severe than the Apollo equipment provides and modification will be required.

S14-008-201	FUEL TRANSFER AND CONDITIONING UNIT PART NO. G14-848030-201

EQUIPMENT SUMMARY DATA SHEET
Sheet 2 of 82

APOLLO FUNCTION SUMMARY

The fuel unit provides storage of the 50/50 UDMH-N_2H_4 during conditioning and prior to delivery to the applicable spacecraft system servicing unit. The unit is mobile and is operationally located in the pad area. It can be operated remotely or locally. The Oxidizer Ready Storage Unit Apollo functions are similar to those of the Fuel Ready Storage Vessel with one exception. The propellant pump capability is 75 gallons per minute (gpm) at a pressure differential of 142 feet of head. The requirements described under unit S14-058 apply to the Oxidizer Ready Storage Vessel as well. The unit is acceptable for use in the oxidizer propellant system of the Voyager Program with modifications.

APOLLO REQUIREMENTS

A. Store fuel (aerozine 50).

B. 3,450 gallon storage capacity at 100 pounds per square inch gauge (psig) maximum.

C. Filtering capacity of 10 micron absolute and 25 micron nominal with a nominal 10 psig across the filter at 75 gpm.

D. Internal pumping capacity of 75 gpm at 180 feet of differential head.

E. Maintain a GN_2 blanket to prevent vaporization.

F. Semi-portable.

G. Interfaces with fuel transfer and conditioning unit and the toxic vapor disposal unit.

VOYAGER REQUIREMENTS

A. 4,760 pounds (approximately 600 gallons) of fuel.

B. Total propellant conditioning and spacecraft propellant servicing time of 16 hours.

SUPPLY REQUIREMENTS

A. 480 volts, 3 phase, 60 cycle, 3 wire.

B. GN_2 at 200 to 300 psig, 7 to 20 standard cubic feet per minute (scfm).

PHYSICAL SIZE

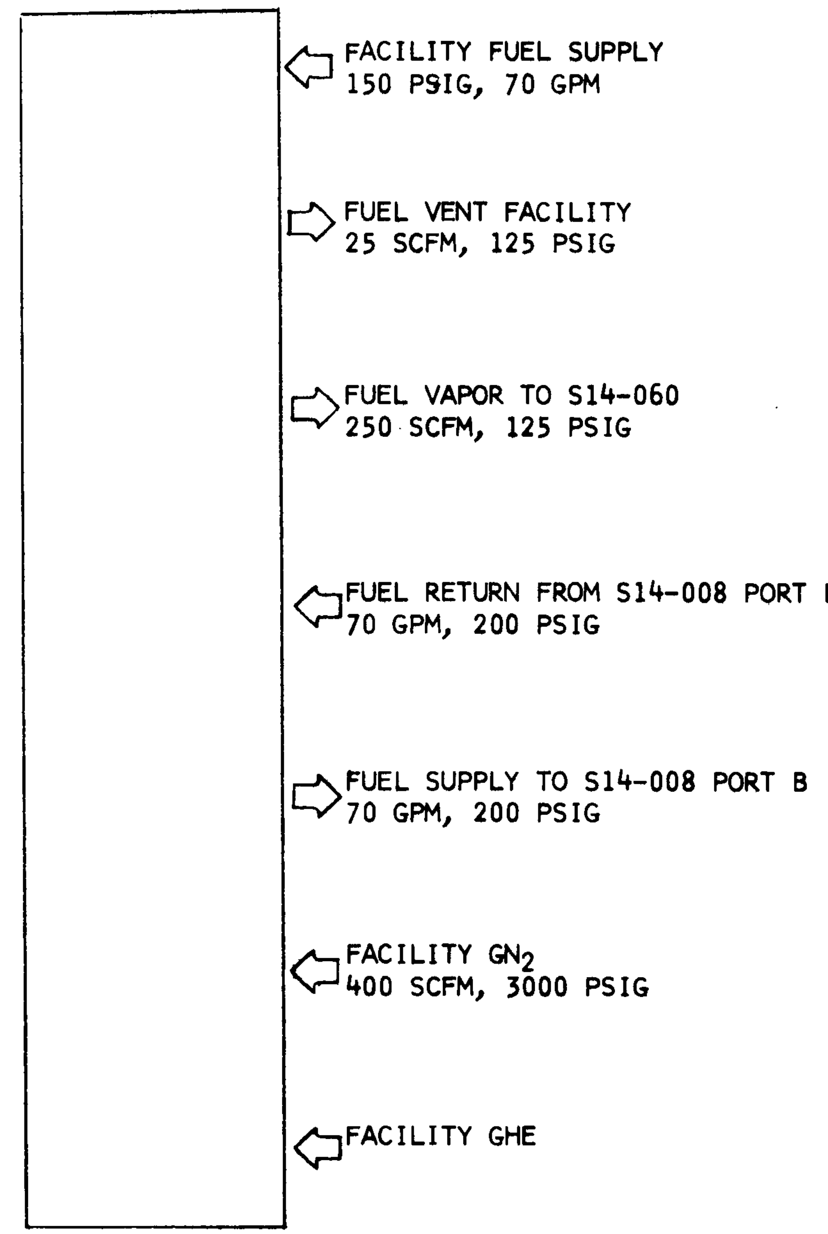
Length - 12 feet.

Width - 8 feet.

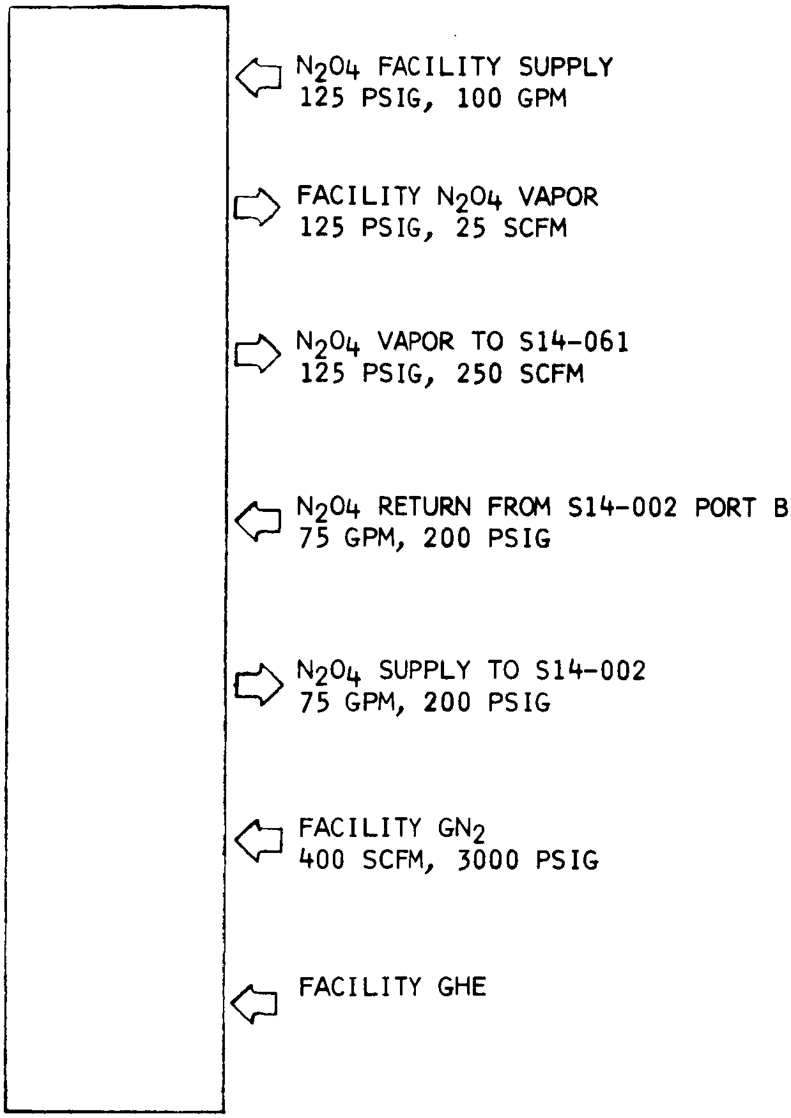
Height - 8 feet.

Weight - 24,000 pounds (dry).

S14-058 FUEL READY STORAGE UNIT



S14-059 OXIDIZER READY STORAGE UNIT



VOYAGER EVALUATION

The Fuel and Oxidizer Ready Storage Units would be applicable for the Voyager Program without modifications.

S14-058-0001	FUEL READY STORAGE UNIT	
	PART NO.	MC 282-0031-1 (NAA)
		LSC 430-94058 (GAEC)
S-14-059	OXIDIZER READY STORAGE VESSEL	
	PART NO.	ME-282-0031-2 (NAA)
		LSC-430-94059

EQUIPMENT SUMMARY DATA SHEET
Sheet 3 of 82

APOLLO FUNCTION SUMMARY

The unit will safely dispose of aerozine 50 vapors generated during the conditioning of the fuel, during nitrogen purge of the system and during the fuel loading operation. The unit is a skid-mounted assembly modular in design, consisting of a gas processing system with necessary instrumentation and controls, capable of either local or remote control of vapor processing procedure control. It is operationally located on the pad area and interfaces with other fuel-handling GSE at the base of the MSS.

APOLLO REQUIREMENTS

The unit will receive propellant vapors mixed with gaseous nitrogen at a minimum flow rate of 100 scfm and will mix the input vapors uniformly with air. The unit will exhaust the air diluted propellant - nitrogen vapors at a minimum flow rate of 180,000 cubic feet per minute with a minimum velocity of 6,000 feet per minute. The unit will be capable of accepting gaseous nitrogen through the vapor inlet port at a rate of up to 400 scfm. The effluent gas shall have a maximum allowable concentration of aerozine 50 of 0.5 ppm exhaust to the atmosphere.

VOYAGER REQUIREMENTS

The assumption has been made that vapors from fluid servicing equipment for Voyager would be exhausted to a facilities drain/vent for processing of the vapors. With this function being performed by an item of GSE, such equipment is required for Voyager. Requirements for Voyager, similar to those specified for this item, were not defined.

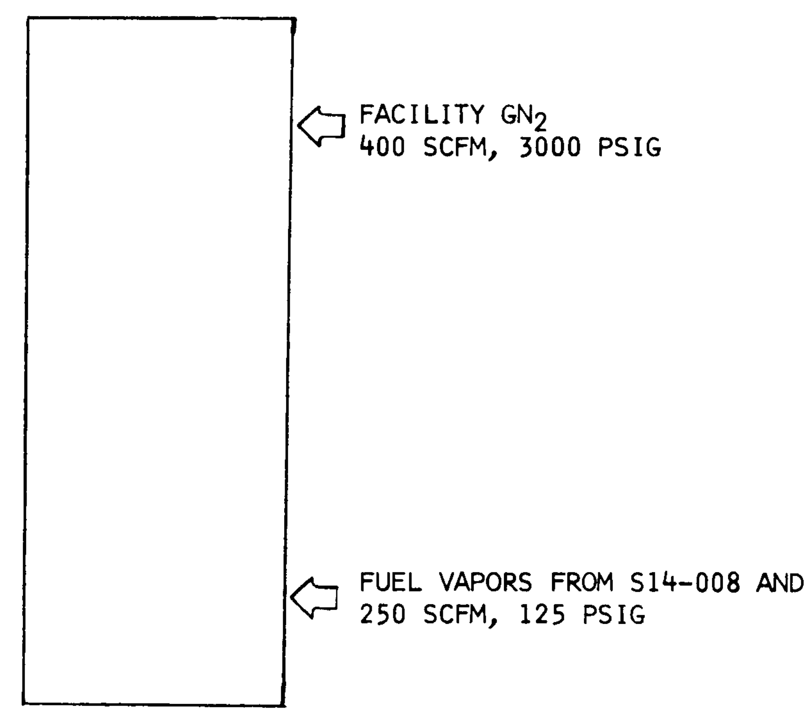
SUPPLY REQUIREMENTS

480/270 volts alternating current (vac), 3 phase, 60 cps.

PHYSICAL SIZE

Length - 8 feet.
Width - 10 feet.
Height - 8 feet.
Weight - 4,000 pounds.

S14-060 FUEL TOXIC VAPOR DISPOSAL UNIT



S14-058

VOYAGER EVALUATION

The propulsion subsystem for Voyager will use MMH or Aerozine 50 (or similar fuel). Because of the long mission life, the allowable leakage will be less than that specified for this item. It is concluded, initially, that this unit can be used for Voyager without modification.

S14-060	VAPOR DISPOSAL UNIT, FUEL (NAA) PART NO. ME 901-0179-0001

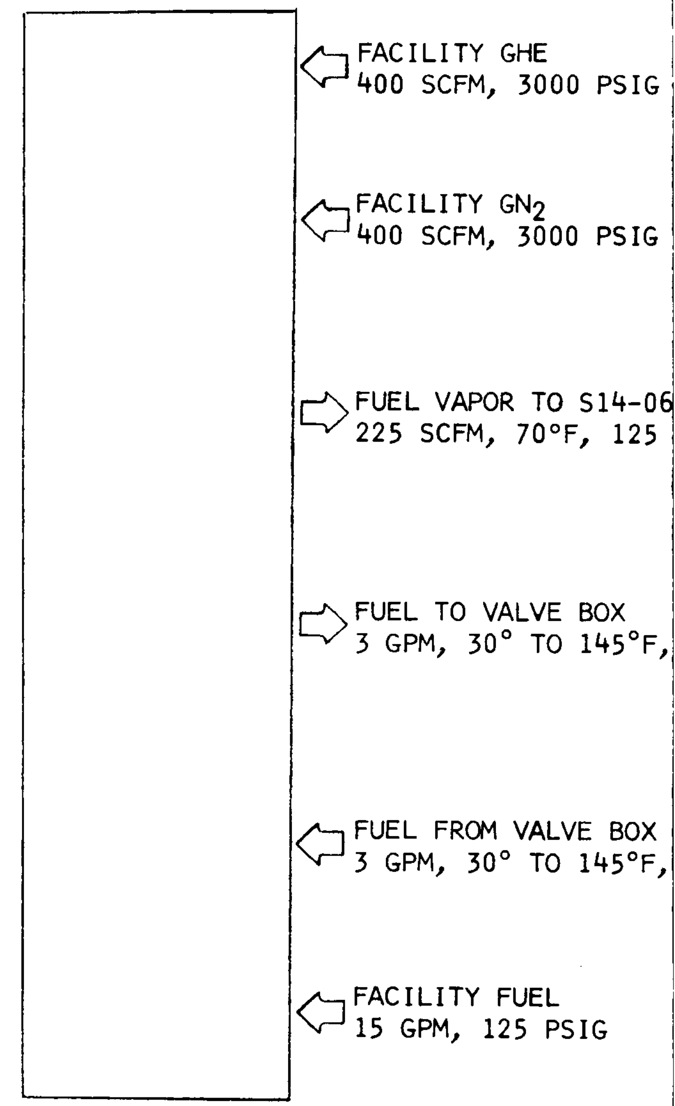
EQUIPMENT SUMMARY DATA SHEET
Sheet 4 of 82

FOLDOUT FRAME

APOLLO FUNCTION SUMMARY

The unit provides a fuel pumping, measuring, evacuation, thermal conditioning, filtering and control system, and a nitrogen pressurization system, totally enclosed in a mobile enclosure. It will deliver 2 ± 1/2 gpm of fuel up to 400 psig and will measure and control the amount of fuel delivered up to 69.7 pounds to each of four service module tanks within ± 1/2 percent of the desired delivery. The unit can condition 800 pounds of fuel in 8 hours or less, evacuate the tanks, and maintain a GN_2 blanket during fill, drain and purge of the RCS tanks. It is a mobile unit located on the "minus 22 foot" level of the service structure and interfaces with the MSS propellant distribution system.

S14-063 S/M RCS FUEL SERVICING UNIT



FOLDOUT FRAME
D-6A

D-6B
FOLDOUT FRAME

0
PSIG

400 PSIG

400 PSIG

VOYAGER EVALUATION

The unit capacity is not sufficient for Voyager Propulsion Subsystem requirements. It can be modified to be more suitable for Voyager and should be considered as a back-up and reviewed later in the program for possible application, if needed.

S14-063 APOLLO S/M RCS FUEL SERVICING
UNIT (NAA)
PART NO. G17-848170

EQUIPMENT SUMMARY DATA SHEET
Sheet 5 of 82

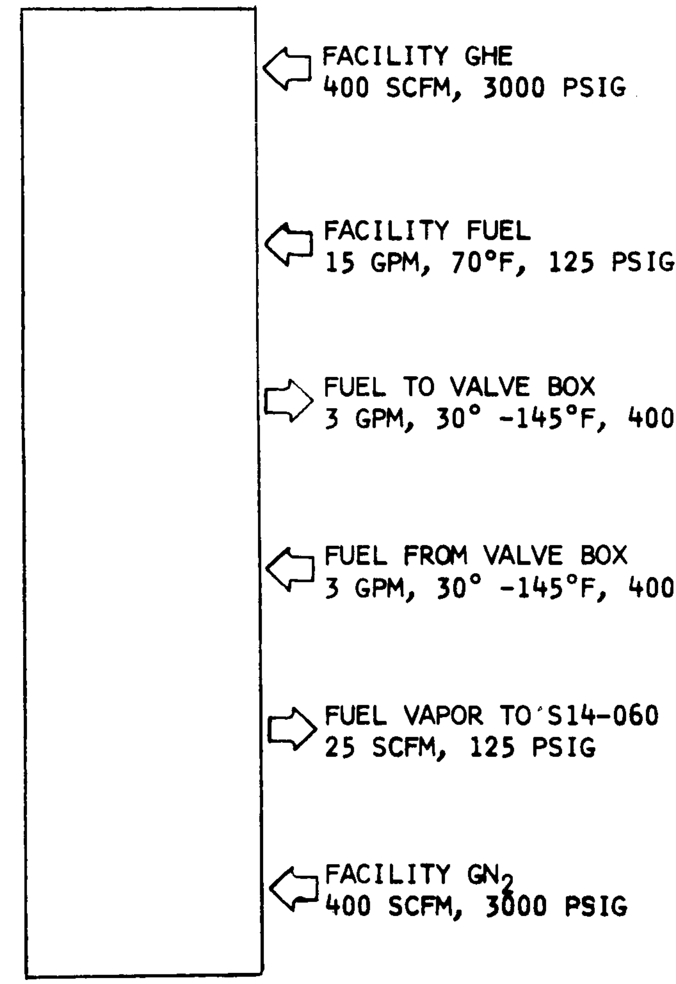
APOLLO FUNCTION SUMMARY

The unit will accept and condition 635 pounds of fuel(MMH), evacuate the C/M RCS fuel tanks at the proper temperature, drain the C/M RCS fuel tanks, and decontaminate and purge the C/M RCS. It is a mobile unit located on the "minus 22 foot" level of the service structure and interfaces with the MSS propellant distribution system.

VOYAGER REQUIREMENTS

None, until the attitude control subsystem propellant is selected. If it is a mono-propellant, this unit may have use on Voyager for loading the A/C tanks and servicing the A/C System.

S14-064 C/M RCS FUEL SERVICING UNI



D-7A

FOLDOUT FRAME

D-7B

FOLDOUT FRAME

PSIG

PSIG

S14-064	COMMAND MODULE RCS FUEL SERVICING UNIT (NAA) PART NO. G16-848380

EQUIPMENT SUMMARY DATA SHEET
Sheet 6 of 82

FOLDOUT FRAME

APOLLO (LEM) FUNCTION SUMMARY

The weigh tank unit is used to calibrate flowmeters and sight gages for propellant loading systems using either live or substitute propellants at the launch pad.

APOLLO (LEM) REQUIREMENTS

A. Weight range of 0 to 4,000 pounds at $\pm$ 0.5 pounds accuracy.
B. Local control and readouts only.
C. Accept fluid at 125 gpm maximum.
D. Discharge fluid at 50 gpm maximum.
E. Pressurize the weigh tank with 50 psig maximum GN_2.

PHYSICAL SIZE

Length - 10 feet.
Width - 7 feet.
Height - 8 feet.
Weight - 4,000 pounds (dry).

VOYAGER REQUIREMENTS

The use of the weigh tank calibration units for Voyager propellant flow-measuring devices checks should be considered. The unit must be modified to provide the accuracy of the Voyager requirements and to adopt flow equipment to this unit for testing.

WEIGH TANK CALIBRATION UNIT (GAEC)	
PART NO.	LDW 430-6190-1 FUEL
	LDW 430-6100-3 OXIDIZER

FUEL VALVE AND CONTROL BOXES (NAA)

The final parts of the fuel distribution system that supplies the propellant to the specific Apollo Spacecraft System consist of the following valve and control boxes:

RCS and SPS Fuel Isolation Valve Box

C/M RCS Fuel Valve Box

S/M RCS Fuel Valve Box

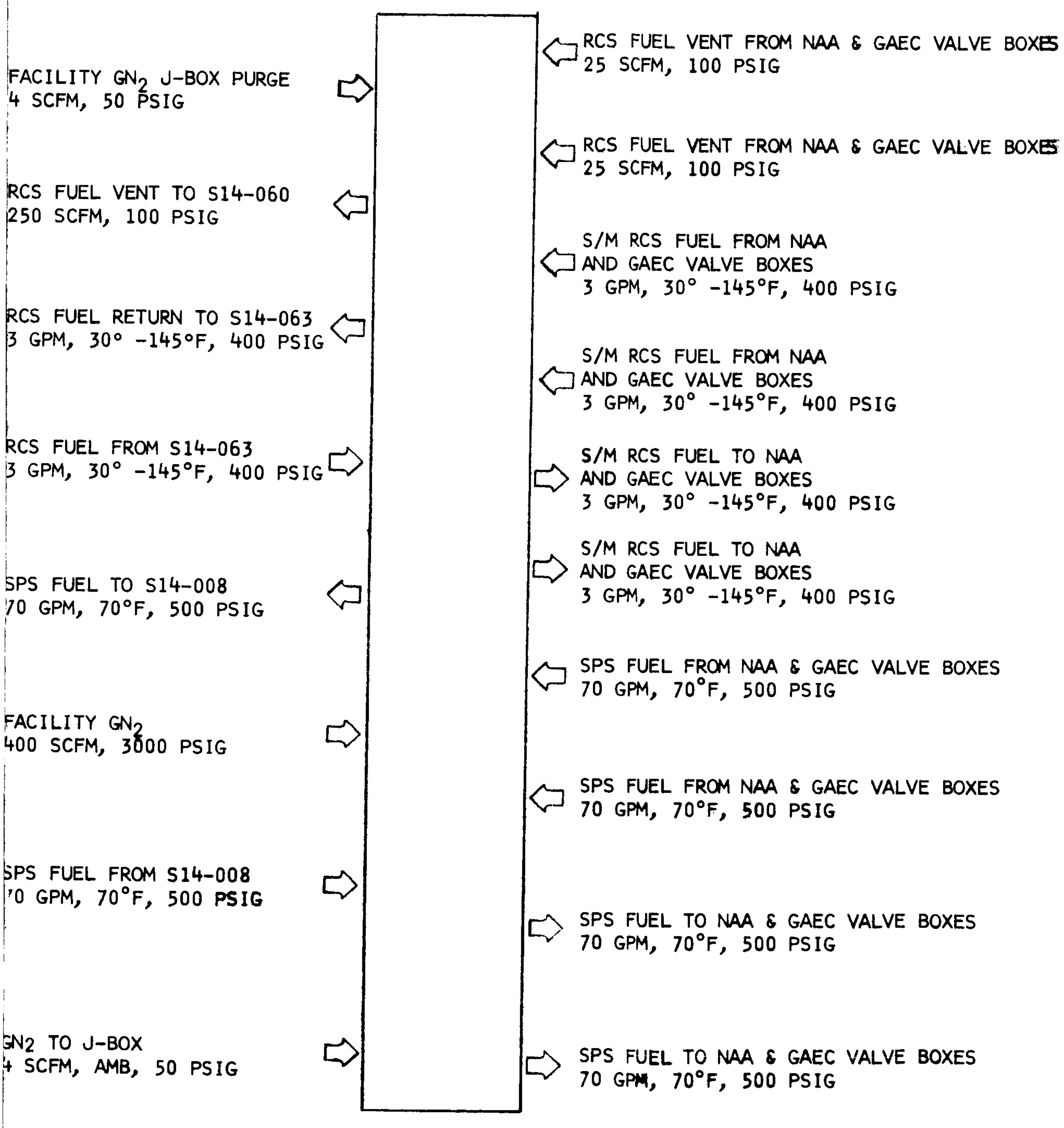
SPS Fuel Valve Box

Fuel Vacuum Unit

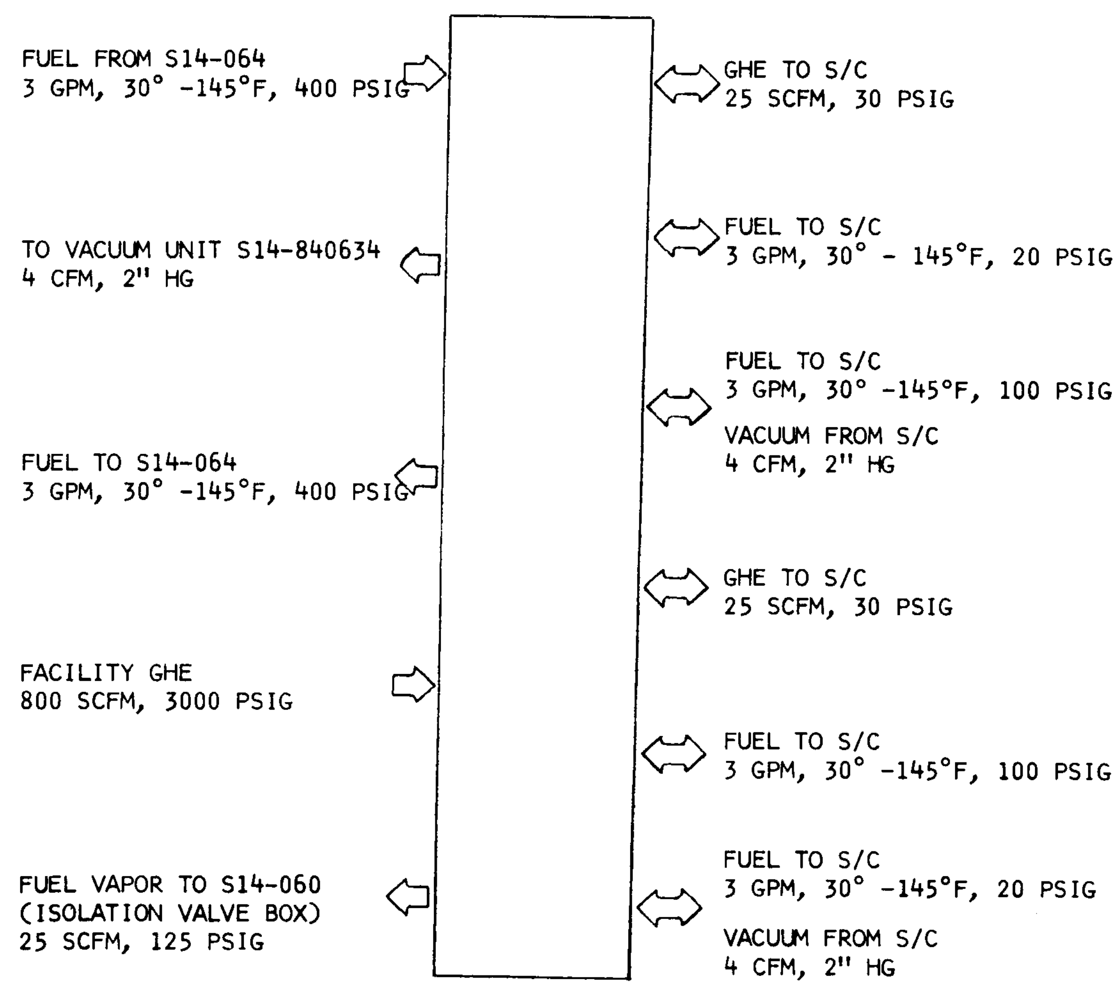
These units, with associated fluid lines and flex hoses, interface with the Spacecraft systems, provide either remote or local control capability, and are the end items for the specialized propellant loading systems. The fluid system capabilities for each item are shown in the following diagrams.

The RCS and SPS Fuel Isolation Valve Box, the SPS Fuel Valve Box, and the Fuel Vacuum Unit, have application to the Voyager program due to a general similarity of fuel requirements. Use of these items in the Voyager Program would depend upon further investigation into the areas of (1) compatibility or modifications for the vehicle interface to the equipment; (2) operational usage and procedural control methods acceptable to the Voyager program; and (3) specific fluid servicing times and deliverable propellant conditions for the spacecraft requirements.

RCS AND SPS FUEL ISOLATION VALVE BOX
8G14-840040



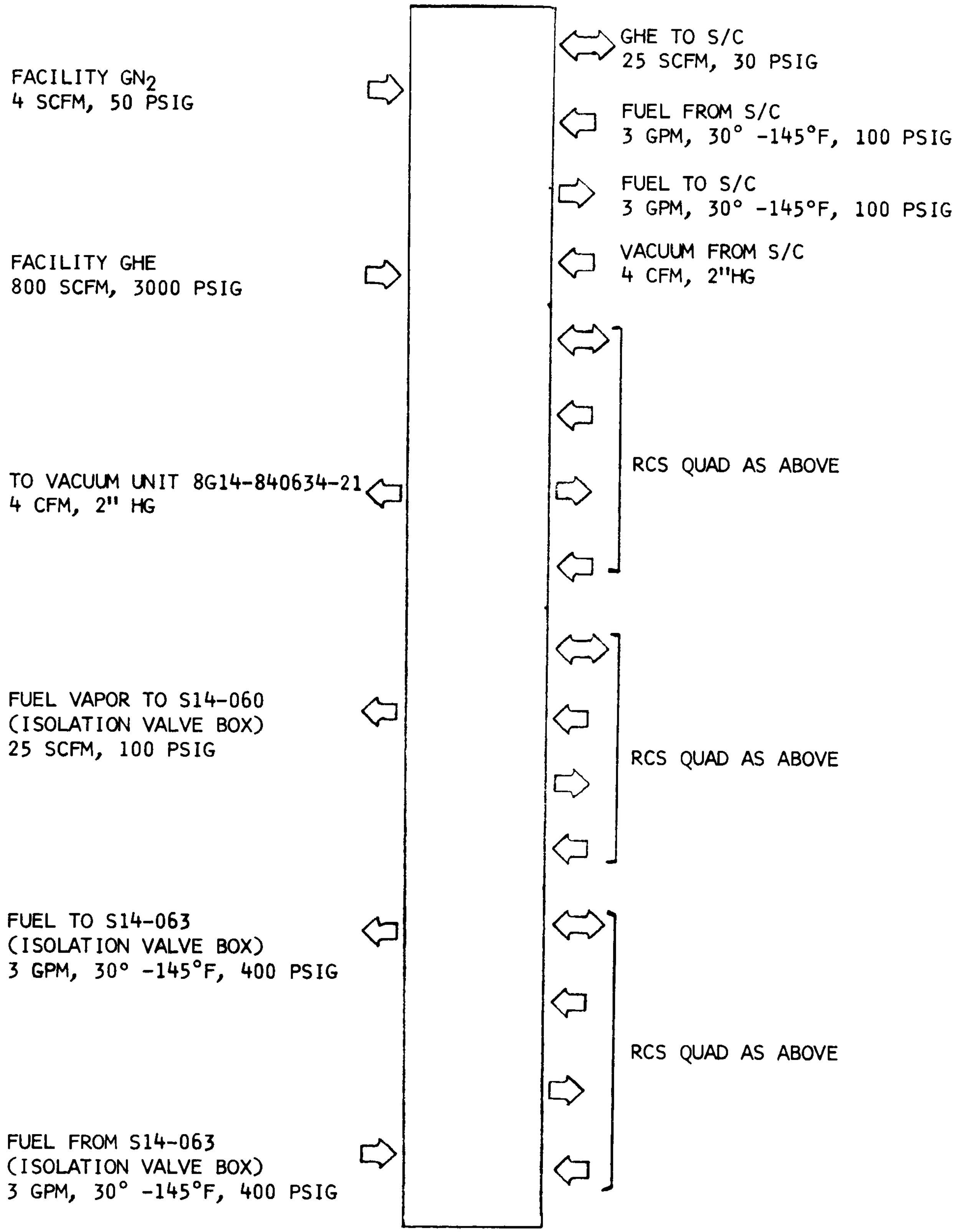
C/M RCS FUEL VALVE BOX



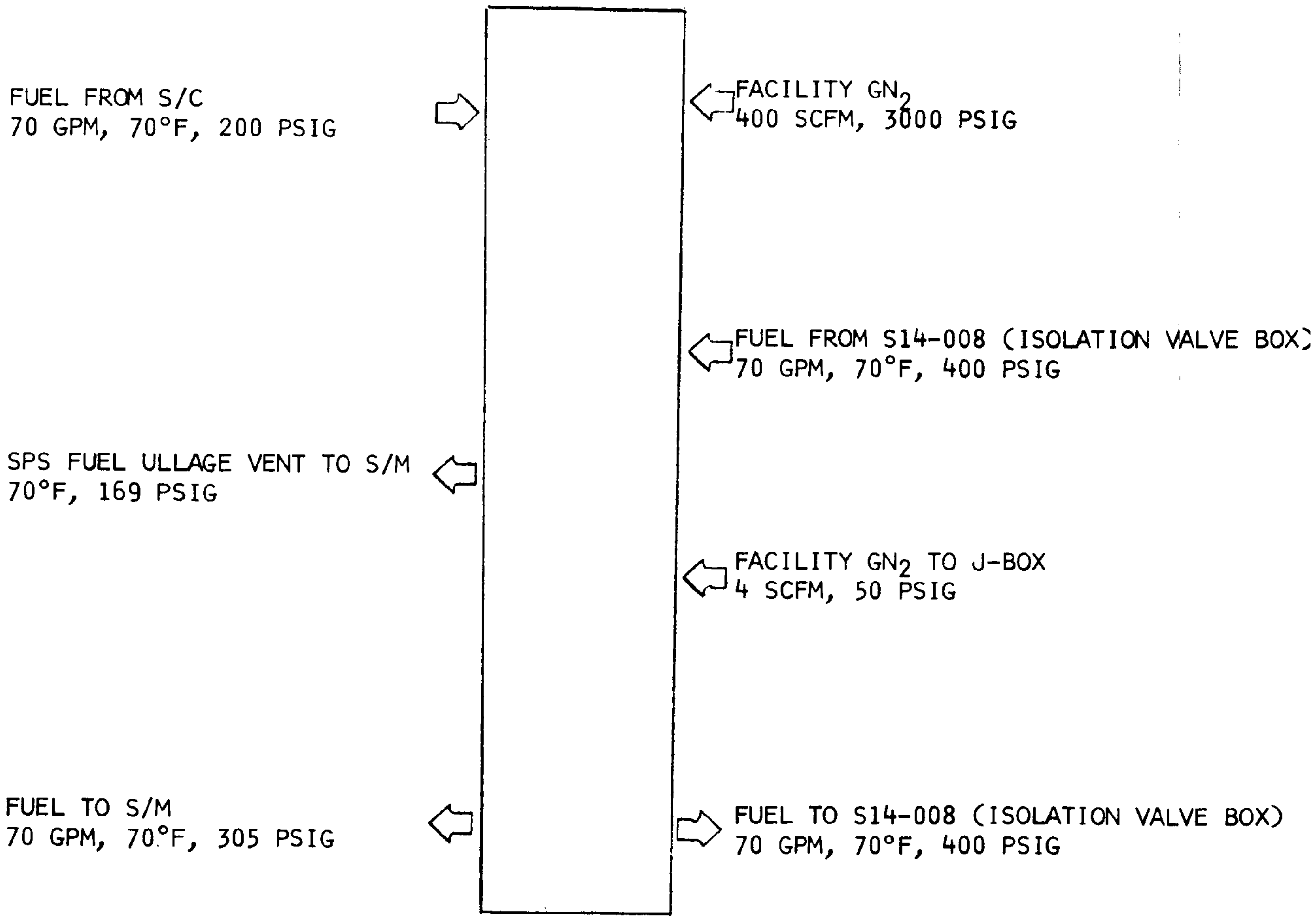
FUEL VALVE AND CONTROL BOXES (NAA)

EQUIPMENT SUMMARY DATA SHEET
Sheet 8A of 82

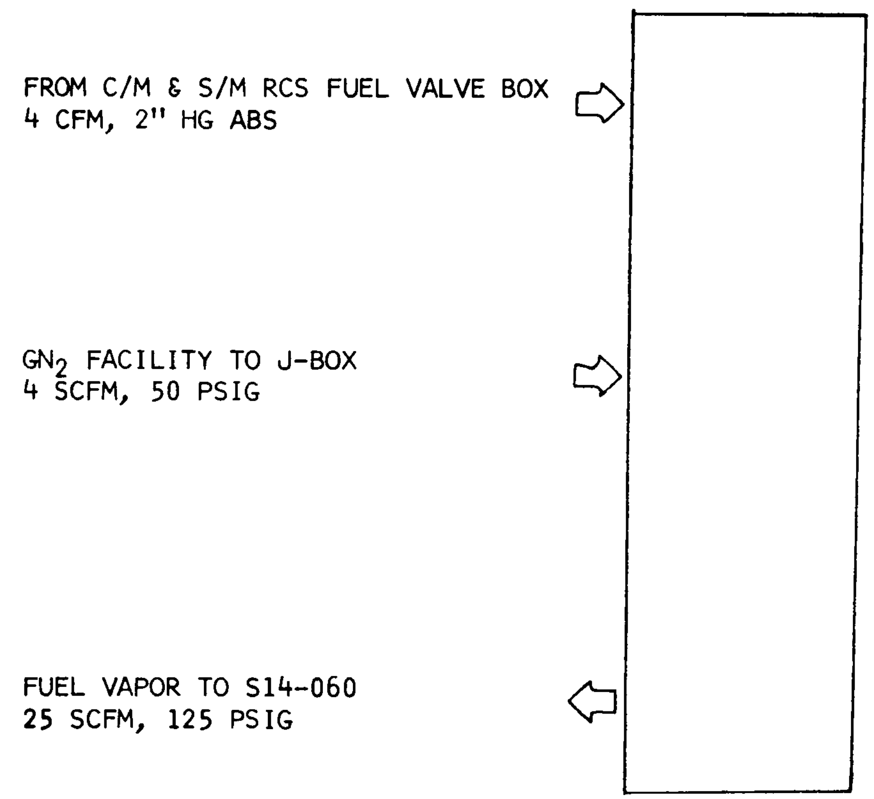
S/M RCS FUEL VALVE BOX



SPS FUEL VALVE BOX
8G14-880070-11



8G14-840634-21 VACUUM UNIT



FUEL VALVE AND CONTROL BOXES (NAA) (CONT)

EQUIPMENT SUMMARY DATA SHEET
Sheet 8B of 82

APOLLO FUNCTION SUMMARY

Transfer predetermined quantities of thermally conditioned and filtered N_2O_4 from ground storage to the Spacecraft (SPS Oxidizer Tanks). The unit is mobile and located on the pad level interfacing with the base of the service structure propellant lines, can be remotely controlled, and has the capability of unloading oxidizer from the Spacecraft and purging the SPS. The unit consists of three major subsystems: a pumping and flow control subsystem, a thermal conditioning subsystem, and a filtering subsystem. Components of these subsystems include pumps, valves, flo meter, pressure and temperature sensors and indicators, micronic filters, heat exchanger, heater and chiller.

APOLLO REQUIREMENTS

A. The unit is capable of cooling 44,000 pounds of propellant from 105 degrees to 70 degrees F. in a period of 8 hours or less when the propellant is gaining heat from an external source at a rate of 100,000 Btu per hour.

B. The unit is capable of heating 44,000 pounds per hour of propellant from 35 degrees F. to 70 degrees F. in a period of 8 hours or less when the propellant is losing heat to an external sink at a rate of 50,000 Btu per hour.

C. The unit is capable of conditioning propellant to any temperature between 35 degrees F. $\pm$ 3 degrees F. and 140 degrees F. $\pm$ 3 degrees F. (The temperature at which the unit is conditioning the propellant will be controlled by a manual temperature controller thermostatic adjustment at the unit.)

D. Measurements accuracies are as follows:

(1) Pressure: Propellant pressure measuring system used in this unit are accurate within 2 percent of the highest pressure indicated by the system.

(2) Temperature: Temperature measuring systems used in this unit are accurate within 1.8 degrees F. between indicated readings of 60 degrees F. to 80 degrees F. When measuring temperatures outside the range of 60 degrees F. to 80 degrees F., the measuring systems are accurate to within $\pm$ 2 percent of the highest temperature indicated by the system.

(3) Flow total and full spacecraft load indicator system: Is accurate to within 0.2 percent of the total mass of propellant leaving the unit up to a maximum total of 30,600 pounds, with minimum delivery of 1,000 pounds.

VOYAGER REQUIREMENTS

A. Oxidizer mass - approximately 7,650 pounds.

B. Temperature range 30 degrees F. to 90 degrees F.

C. Accuracy of temperature measurement - 2 1/2 degrees F.

D. Loading time, including all connections, calibrations, purging, evacuating, filling and disconnecting, estimated to be 16 hours.

E. Accuracy of oxidizer weight measurement is within 0.07 percent of full load.

SUPPLY REQUIREMENTS

GN_2, 500 to 3,000 psig, up to 400 scfm flow at ambient temperature.

Electrical - 480 vac, 3 phase, 60 cps, 166 kva.

PHYSICAL SIZE

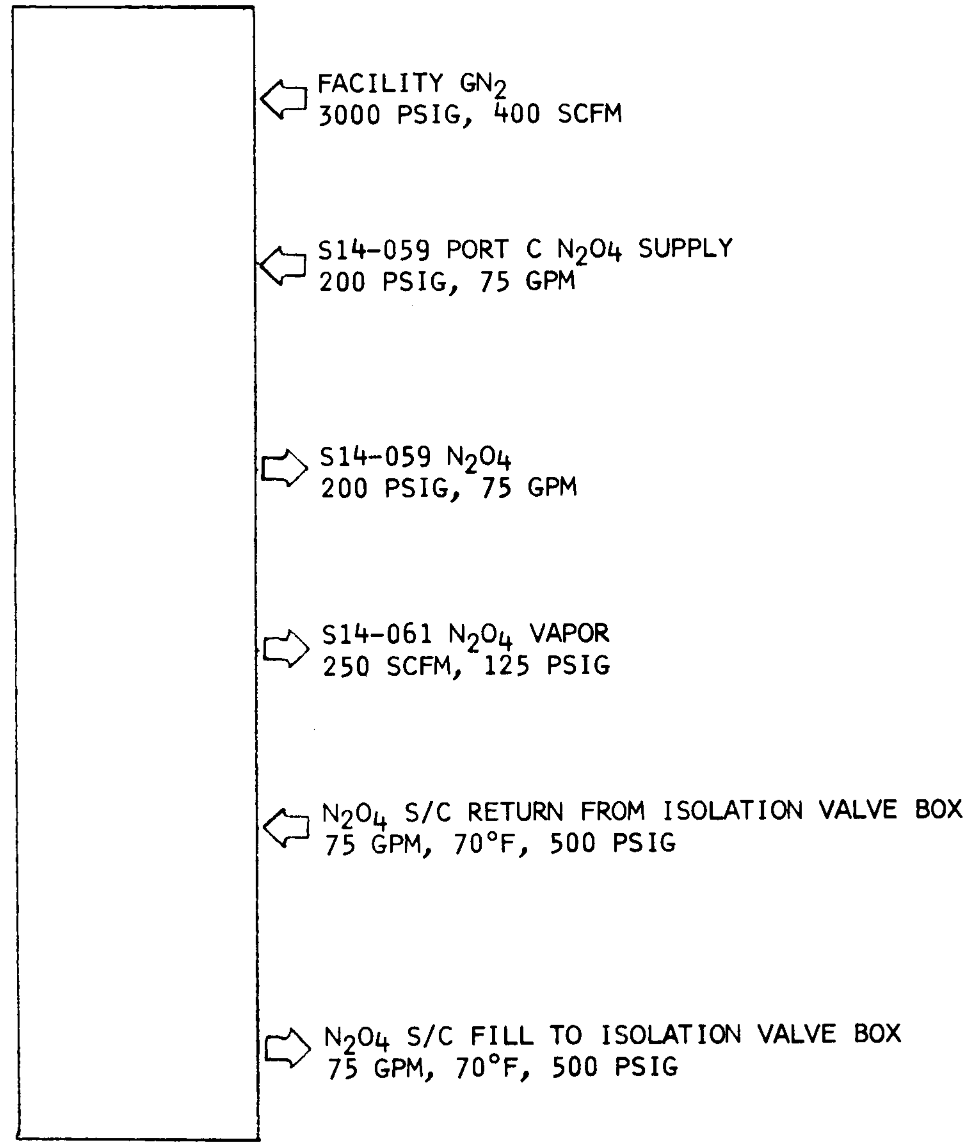
Length - 250 inches.

Width - 150 inches.

Height - 131 inches.

Weight - Approximately 23,000 pounds.

S14-002 OXIDIZER TRANSFER AND CONDITIONING UNIT



VOYAGER EVALUATION

The Apollo Oxidizer Transfer Conditioning Unit is applicable for use on the Voyager Program. The weight of propellant required and the conditions under which it is furnished are compatible. The accuracy of weight measurement for Voyager is more severe and modification of the Apollo unit would be required.

S14-002-201	OXIDIZER TRANSFER/CONDITIONING UNIT PART NO. G14-848029-201

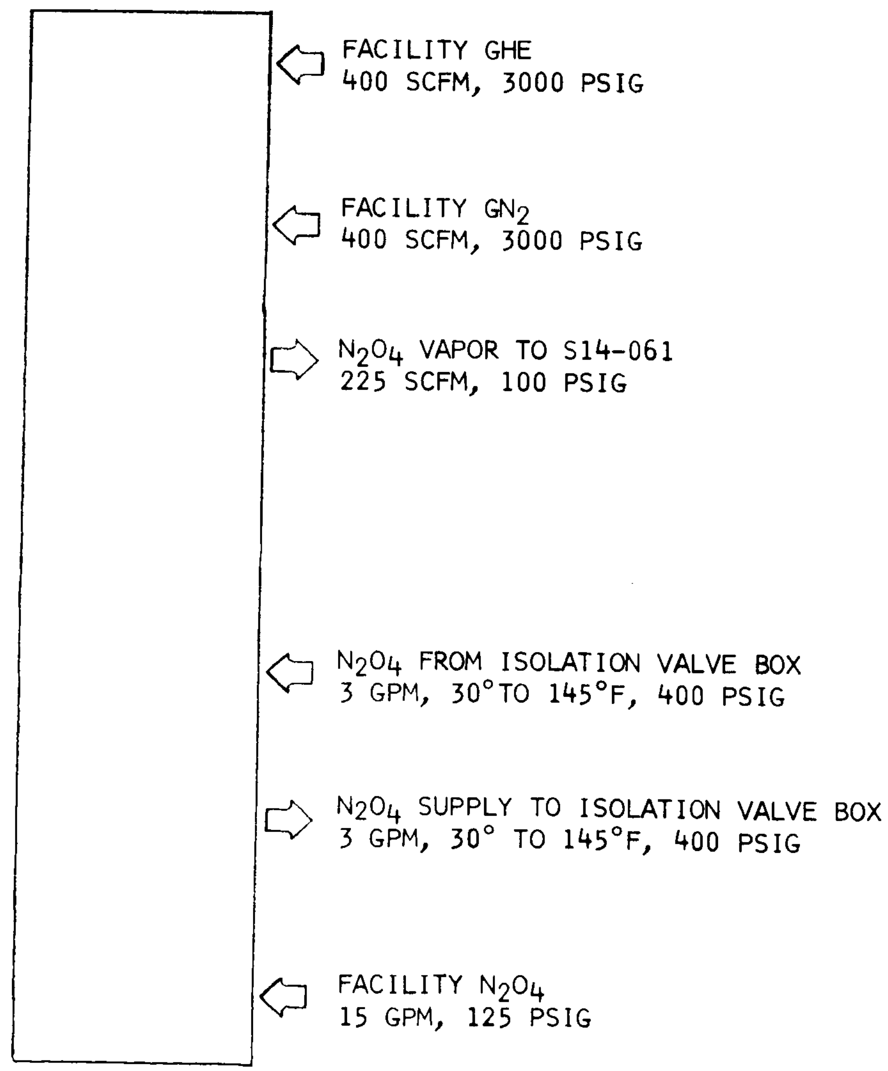
EQUIPMENT SUMMARY DATA SHEET

Sheet 9 of 82

APOLLO FUNCTION SUMMARY

The unit provides an oxidizer pumping, measuring, evacuation, thermal conditioning, filtering and control system, and a nitrogen pressurization system totally encased in a mobile enclosure. It is located on the "minus 22 foot" level of the service structure and interfaces with the MSS propellant distribution system. It will deliver 2 ± 1/2 gpm of oxidizer at 400 psig, and will measure and control the amount of oxidizer delivered up to 138.1 pounds in each of four service module tanks and 89.2 pounds to each of two command module tanks. Total oxidizer handled is 2,540 pounds. It can evacuate the tanks, maintain a GN_2 blanket during fill, drain and purge the RCS tanks.

S14-057 RCS OXIDIZER SERVICING UNIT



VOYAGER EVALUATION

This unit capability is not sufficient for Voyager Propulsion subsystem requirements. It can be modified to be more suitable for Voyager and should be considered as a back-up and reviewed later in the program for possible application, if needed.

S14-057	APOLLO RCS OXIDIZER SERVICING UNIT (NAA) PART NO. G14-848380

EQUIPMENT SUMMARY DATA SHEET
Sheet 10 of 82

APOLLO FUNCTION SUMMARY

The unit will safely dispose of N_2O_4 vapors generated during the conditioning of the oxidizer, during nitrogen purge of the system, and during the oxidizer loading operation. This is a skid-mounted assembly modular in design, consisting of a gas-processing system with necessary instrumentation and controls, capable of either local or remote control. It is operationally located on the pad and interfaces with other oxidizer-handling GSE at the base of the MSS.

APOLLO REQUIREMENTS

The unit will receive propellant vapors mixed with gaseous nitrogen at a minimum flow rate of 100 scfm and will mix the input vapors uniformly with air. The unit will exhaust the air diluted propellant - nitrogen vapors at a minimum flow rate of 180,000 cubic feet per minute with a minimum velocity of 6,000 feet per minute. The unit shall be capable of accepting gaseous nitrogen through the vapor inlet port at a rate of up to 400 scfm. The effluent gas shall have a maximum allowable concentration of 5 ppm of NO_2 and 2.5 ppm N_2O_4 exhausted to the atmosphere.

VOYAGER REQUIREMENTS

The assumption had been made that fluid servicing equipment for Voyager would be exhausted to a facilities drain or vent for processing of the vapors. With this function being performed by an item of GSE, such equipment is required for Voyager. Requirements for Voyager, similar to those specified for this item, were not defined.

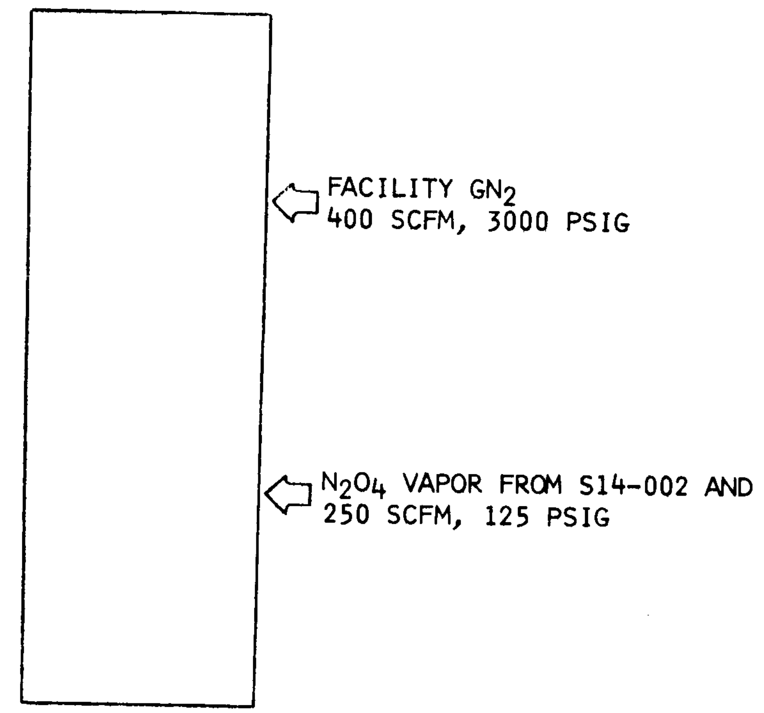
SUPPLY REQUIREMENTS

480/270 vac, 3 phase, 4 wire, 60 cps.

PHYSICAL SIZE

Length - 8 feet.
Width - 10 feet.
Height - 8 feet.
Weight - 4,000 pounds.

S14-061 OXIDIZER TOXIC VAPOR DISPOSAL U



NIT

S14-059

VOYAGER EVALUATION

The propulsion subsystem for Voyager will use N_2O_4 oxidizer. Because of the long mission life, the allowable leakage will be less than that specified for Apollo. It is concluded, initially, that this unit can be used for Voyager without modification.

S14-061 VAPOR DISPOSAL UNIT, OXIDIZER (NAA)
PART NO. ME 901-0178-0001

EQUIPMENT SUMMARY DATA SHEET
Sheet 11 of 82

APOLLO FUNCTION SUMMARY

This unit is used to receive oxidizer bled from the SPS and provide a quantity measurement of the amount received. The unit is portable, is operationally located on platform IVA of the service structure, and interfaces with the MSS propellant system.

APOLLO REQUIREMENTS

The equipment shall be capable of accepting oxidizer from the SPS (drained into the 30 gallon capacity tank). Flow is visually displayed. The tank shall be drained by flow of GN_2 into the tank. The GN_2 also acts as a purge. This unit is compatible with ACE.

VOYAGER REQUIREMENTS

Current planning for loading fuel and oxidizer into the Voyager Spacecraft does not utilize the method of filling the tanks full and then "off-loading" a measured amount. Based on those requirements, this unit is not usable for Voyager. However, if the loading method changes, this item should be considered for Voyager use.

VOYAGER EVALUATION

This unit can be used to pressurize the fuel and oxidizer tanks of the propulsion subsystem, and the tanks of the attitude control subsystem during transport of the assembled spacecraft.

S14-122	SERVICE PROPULSION SYSTEM OXIDIZER BLEED MEASURING EQUIPMENT (NAA) PART NO. G14-849850-101

EQUIPMENT SUMMARY DATA SHEET
Sheet 12 of 82

OXIDIZER VALVE AND CONTROL BOXES (NAA)

The final parts of the oxidizer distribution system that supplies propellant to the specific Apollo Spacecraft System consist of the following valve and control boxes:

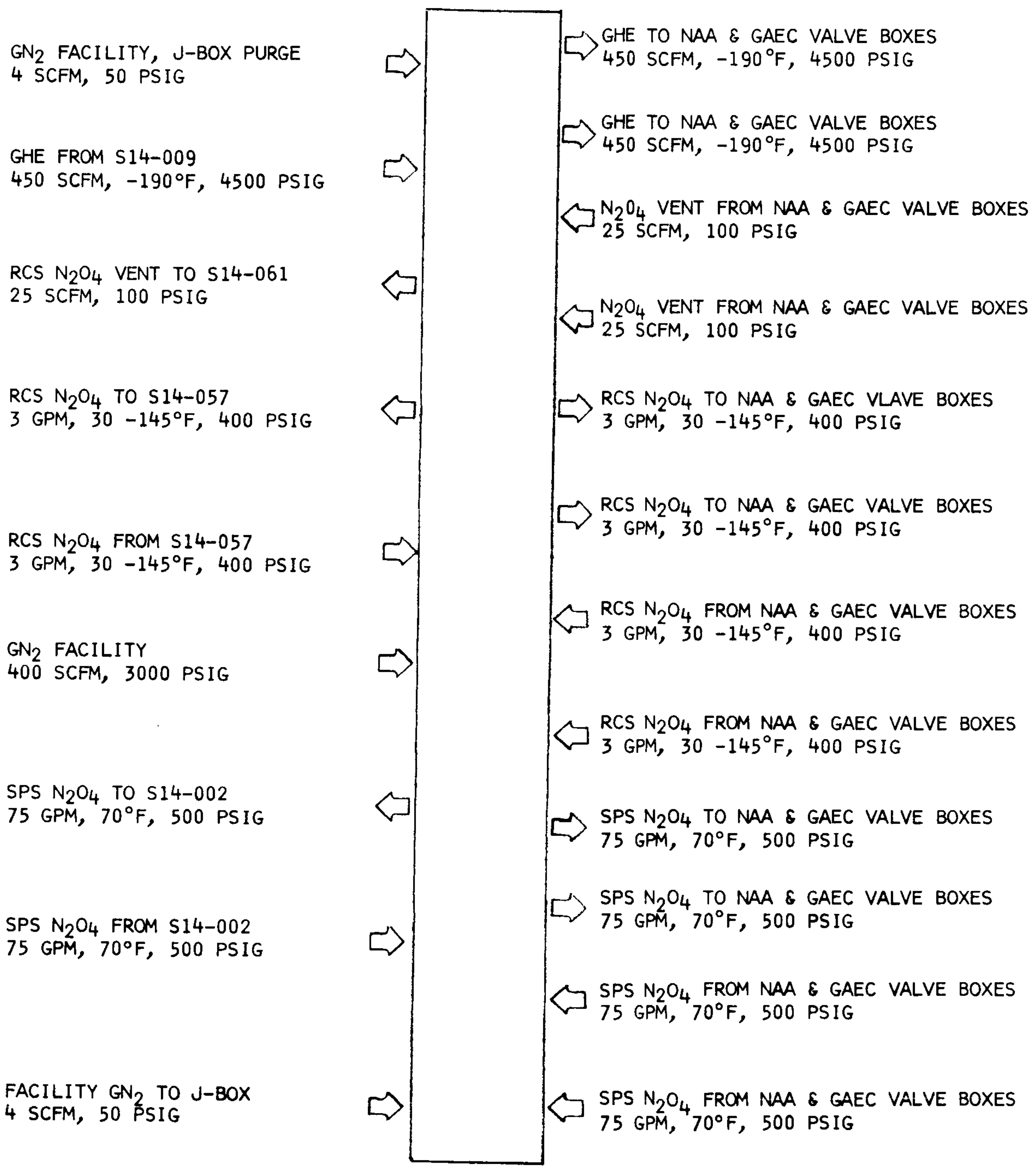
Oxidizer/Helium Isolation Valve Box
C/M RCS Oxidizer Valve Box
S/M RCS Oxidizer Valve Box
SPS Oxidizer Valve Box
Oxidizer Vacuum Unit

These units, with associated fluid lines and flex hoses, interface with the spacecraft systems; provide either remote or local control capability; and are the end items for the specialized propellant loading systems. The fluid system capabilities for each item are shown in the following diagrams.

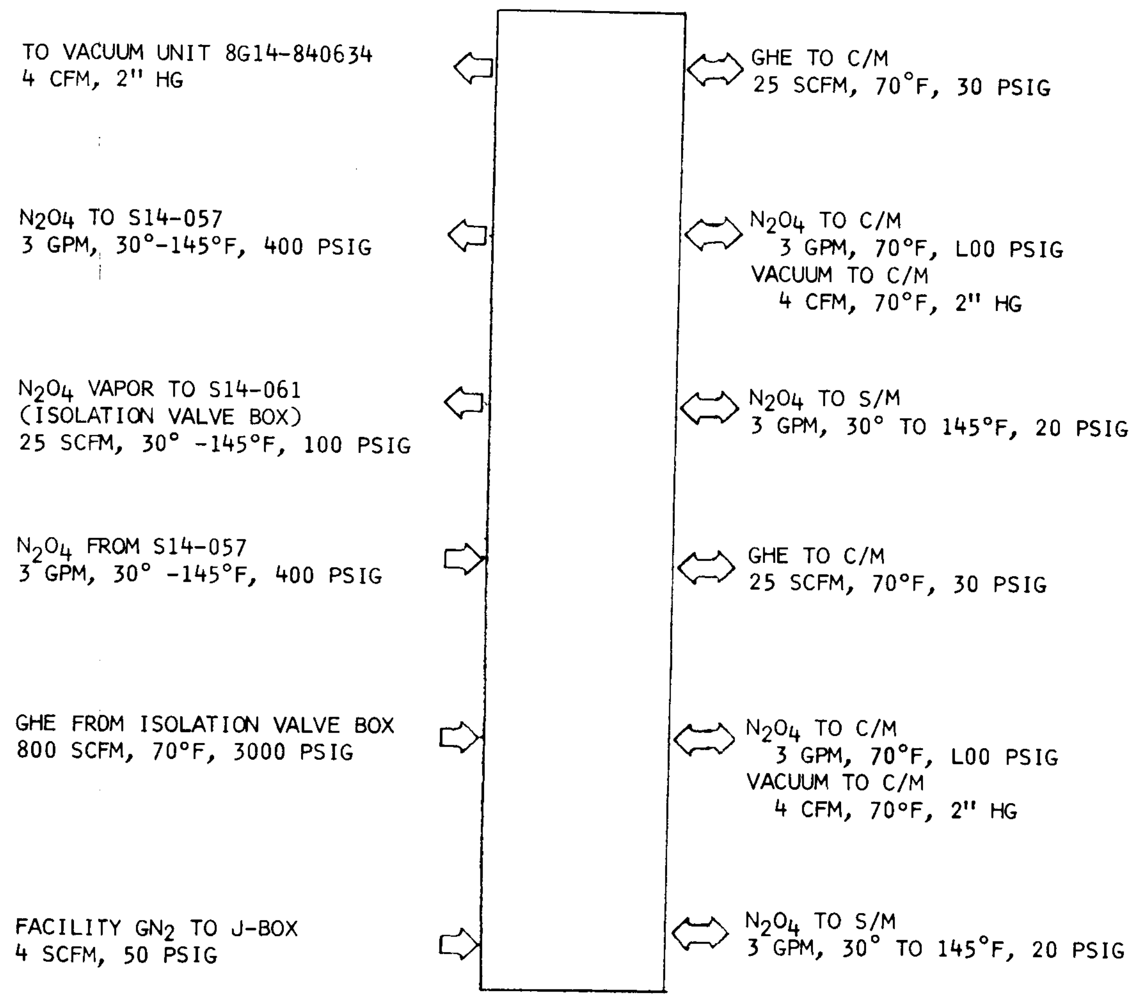
The Oxidizer/Helium Isolation Valve Box, the SPS Oxidizer Valve Box, and the Oxidizer Vacuum Unit have application to the Voyager Program due to a general similarity of oxidizer requirements. Use of these items in the Voyager Program would depend upon further investigation into the areas of (1) compatibility and modifications for the vehicle interface to the equipment; (2) operational usage and procedural control methods acceptable to the Voyager Program; and (3) specific fluid servicing times and deliverable propellant conditions for the spacecraft requirements.

FOLDOUT FRAME

OXIDIZER/HELIUM ISOLATION VALVE BOX
8G14-840041



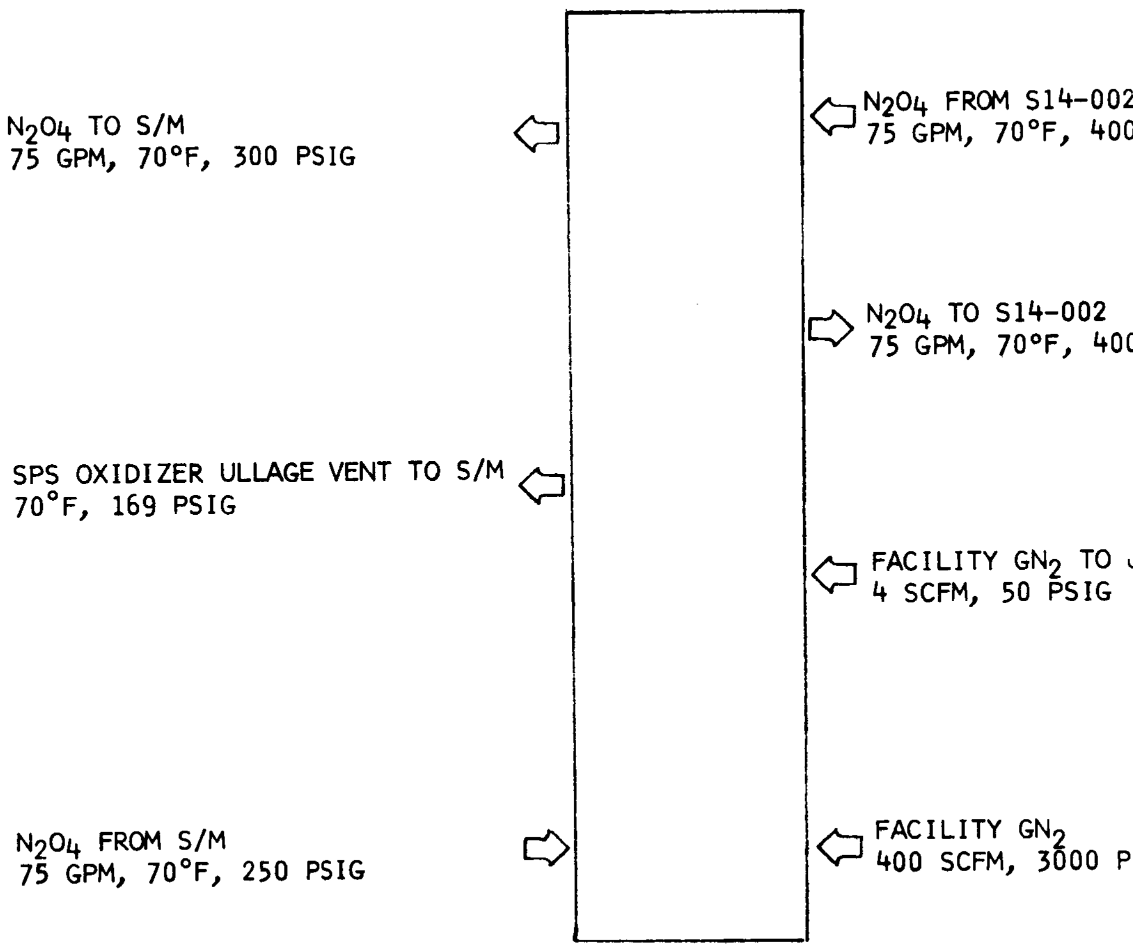
C/M RCS OXIDIZER VALVE BOX
8G14-840232-21



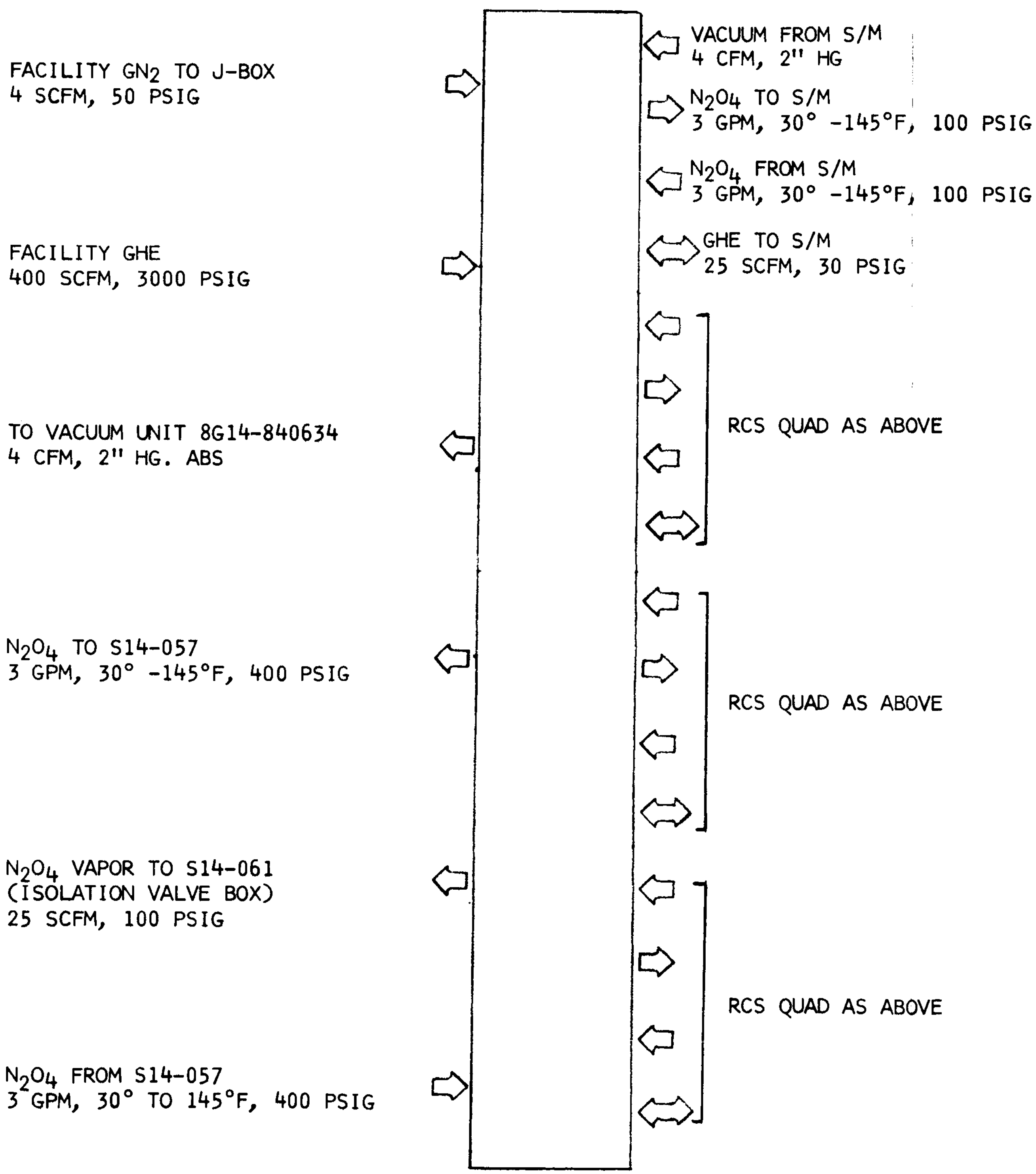
OXIDIZER VALVE AND CONTROL BOXES (NAA)

EQUIPMENT SUMMARY DATA SHEET
Sheet 13A of 82

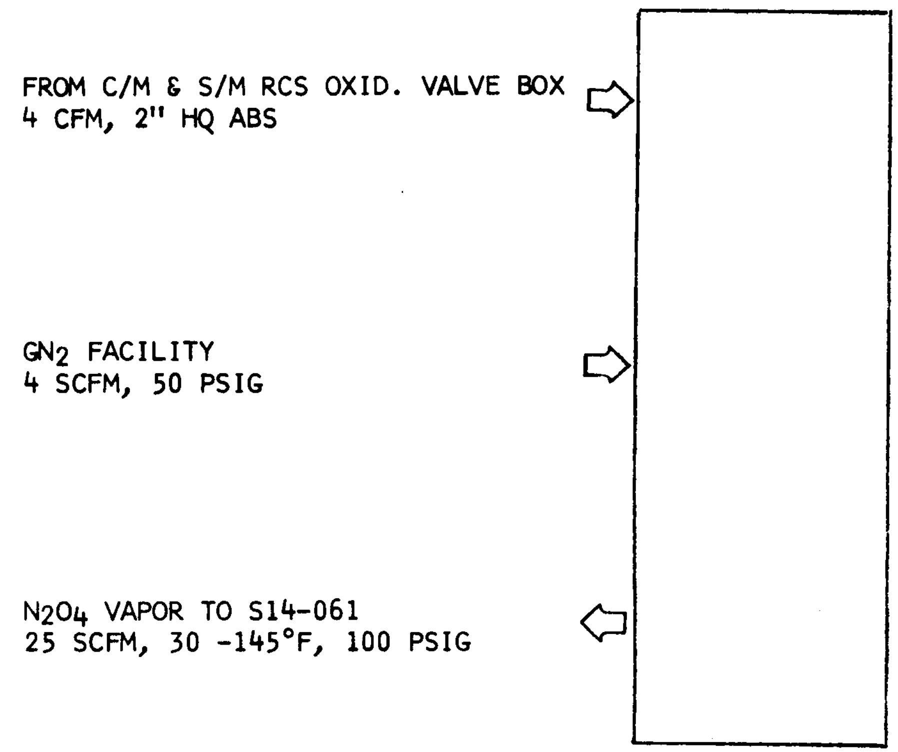
SPS OXIDIZER VALVE BOX
8G14-880070-21



S/M RCS OXIDIZER VALVE BOX
8G14-840001-41



8G14-840634-11 VACUUM UNIT



OXIDIZER VALVE AND CONTROL BOXES (NAA) (CONT)

EQUIPMENT SUMMARY DATA SHEET
Sheet 13B of 82

2.0 GAS HANDLING EQUIPMENT

2.1 Helium

Helium Transfer and Conditioning Unit, S14-009 (NAA), LSC-430-94009 (GAEC)

Helium Booster Unit, S14-022 (NAA), LSC-430-94033 (GAEC)

Helium Ready Storage Container, S14-062-0001 (NAA), LSC-430-94062 (GAEC)

Helium Valve and Control Boxes (NAA)

2.2 Nitrogen

Protective Pressurization Unit Assembly, S14-099 (NAA)

APOLLO FUNCTION SUMMARY

Transfers and filters gaseous helium at controlled pressures and temperatures from ground storage (unit S14-062) to spacecraft propellant helium storage tanks. Unit is portable, located on the "minus 22 foot" level of the service structure, interfaces with the service structure propellant systems, can be remotely controlled, and has the capability of depressurizing the onboard helium storage tanks.

APOLLO REQUIREMENTS

A. Transfer gaseous helium at 6,000 psig from ground storage unit to spacecraft systems at 4,500 and 3,500 psig.

B. Condition helium with liquid nitrogen heat exchanger to range of minus 190 degrees F. to plus 70 degrees F.

C. Filter helium to 10 microns nominal.

D. Automatic shutoff, bleed, and topping off at desired pressure settings.

E. Gaseous helium flow capability of 800 scfm maximum.

F. Mobility.

VOYAGER REQUIREMENTS

Helium (gaseous) will be required for propulsion and attitude control system tank pressurizations.

SUPPLY REQUIREMENTS

A. 120 vac 60 cps single phase.

B. 28 volts dc.

C. LN_2 (100 gallons capacity).

D. Gaseous helium at 6,000 psig, 100 degrees F.

E. GN_2 at 100 to 3,000 psig range ambient temperature with 10 scfm maximum.

PHYSICAL SIZE

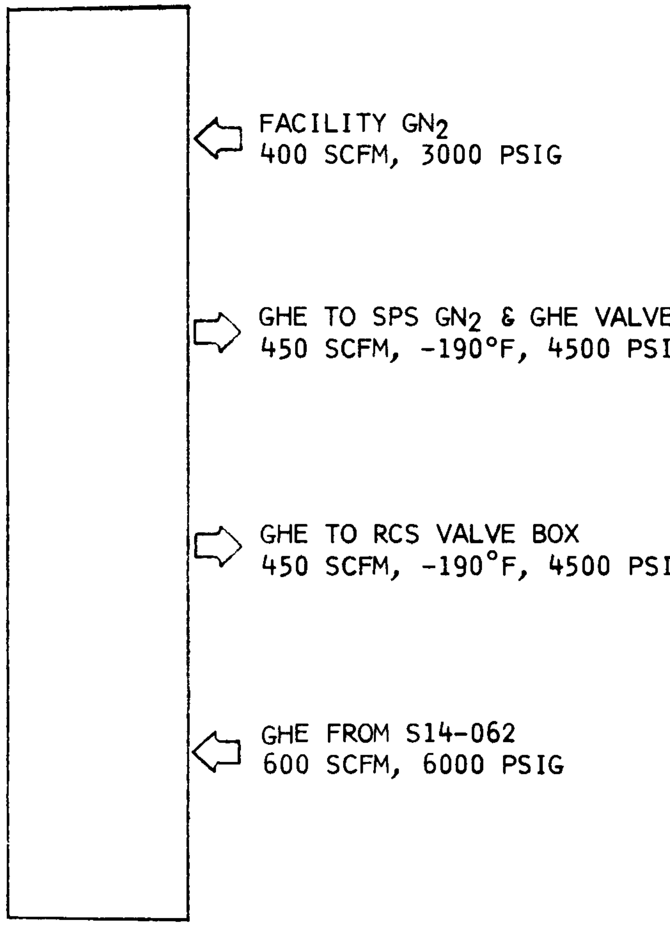
Length - 119 inches.

Width - 73.5 inches.

Height - 82 inches.

Weight - 3,800 pounds (wet).

S14-009 HELIUM TRANSFER UNIT



BOX
G

G

VOYAGER EVALUATION

The helium transfer and conditioner unit would be applicable for Voyager. Modification requirements to be considered would be flow capacities, filtering, and temperature conditioning applications.

S14-009	HELIUM TRANSFER AND CONDITIONER UNIT	
	PART NO.	MC-901-0123-0001 (NAA)
		LSC-430-94009 (GAEC)

EQUIPMENT SUMMARY DATA SHEET
Sheet 14 of 82

APOLLO FUNCTION SUMMARY

The unit filters, dries, and transfers gaseous helium at controlled temperatures and pressures from ground storage to GSE storage units. The unit is portable, located on the pad, and interfaces with the service structure fluid distribution system.

APOLLO REQUIREMENTS

A. Deliver 50 scfm at 6,000 psig.
B. Condition for helium outlet temperature of 100 degrees F.
C. Filter to 10 micron absolute.
D. Outlet dew point of minus 08 degrees F.
E. Pump down LEM helium tanks to 300 psig during detanking.

VOYAGER REQUIREMENTS

Helium pressurization will be required for propulsion propellant tank pressurization (approximately 70 pounds).

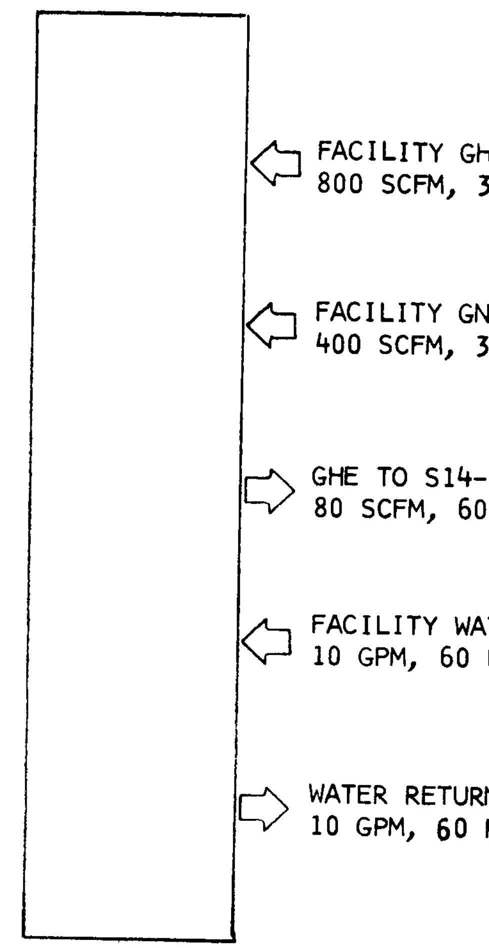
SUPPLY REQUIREMENTS

A. 220/480, 3 phase, 4 wire.
B. Cooling water at 20 gpm.
C. Helium from storage at 100 to 2,500 psig.

PHYSICAL SIZE

Length - 60 inches.
Width - 100 inches.
Height - 60 inches.
Weight - 3,000 pounds.

S14-022 HELIUM BOOSTE



R UNIT

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TER SUPPLY
PSIG

N TO FACILITY
PSIG

VOYAGER EVALUATION

The Helium Booster Unit is applicable for use on the Voyager Program. Facility helium supply is also available at 2,200 psig and this unit would be used to boost the pressure to 6,000 psig if required. The unit would not require modifications and would be compatible with Voyager requirements.

S14-022	HELIUM BOOSTER UNIT	
	PART NO.	MC-901-0101 (NAA)
		LSC-430-94022 (GAEC)

EQUIPMENT SUMMARY DATA SHEET
Sheet 15 of 82

FOLDOUT FRAME

APOLLO FUNCTION SUMMARY

This unit is for storage at the pad of high pressure helium for pressurization of the spacecraft propellant systems.

APOLLO REQUIREMENTS

A. Store helium at the pad at 6,000 psig in sufficient quantity to provide a blowdown capacity of 120 pounds of helium before a storage decay below 5,000 psig.

B. Provide two helium fillings of the propulsion helium tanks to 4,500 psig and the RCS tanks to 3,500 psig.

C. Unit must be transportable between the facility supply and the launch complex site.

D. Delivery flow rate must be 8 pounds per minute.

E. Unit shall filter all particles over 25 microns and 98 percent of all 10 micron or greater particles.

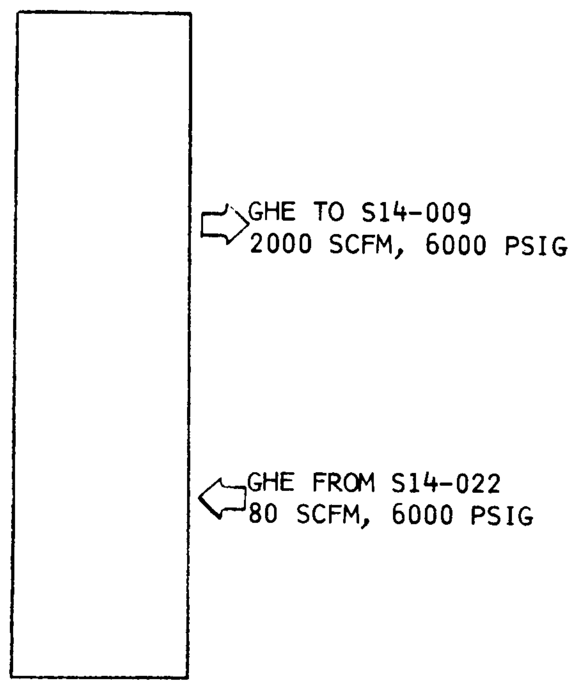
VOYAGER REQUIREMENTS

Seventy pounds of helium will be required for the propulsion system tank pressurizations. The propulsion subsystem pressurant tank pressure is 3,600 psig.

PHYSICAL SIZE

Length - 8 feet.
Width - 32 feet.
Height - 8 feet.
Weight - 40,000 pounds.
Storage Volume - 250 cubic feet (water volume).

S14-062 HELIUM STORAGE UNIT



VOYAGER EVALUATION

The helium storage units would be applicable for use on the Voyager Program without modifications.

S14-062-0001	HELIUM READY STORAGE CONTAINER	
	PART NO.	MC-183-0018 (NAA)
		LSC-430-94062 (GAEC)

EQUIPMENT SUMMARY DATA SHEET
Sheet 16 of 82

FOLDOUT FRAME

HELIUM VALVE/CONTROL BOXES

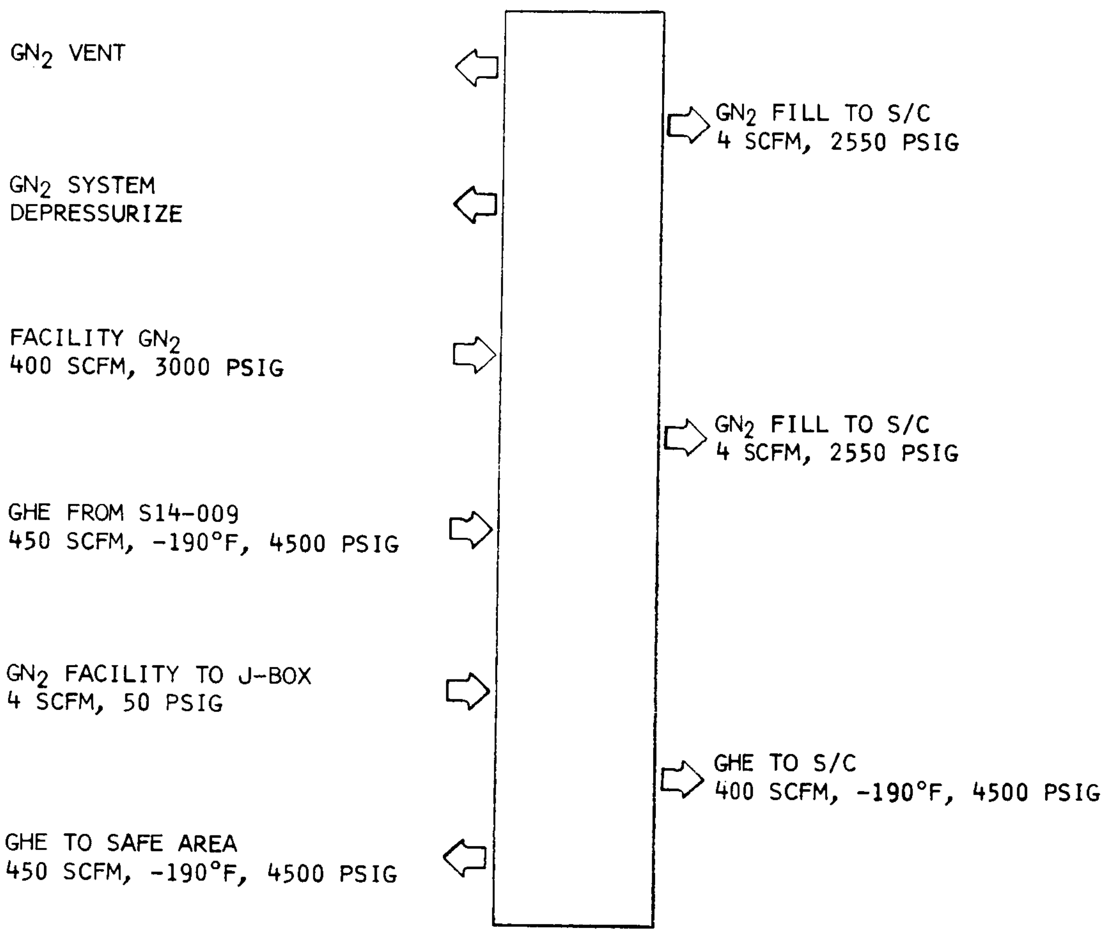
The final parts of the gaseous helium and nitrogen distribution system that supplies gas to the specific Apollo Spacecraft System consist of the following valve and control boxes:

RCS Helium Valve Box

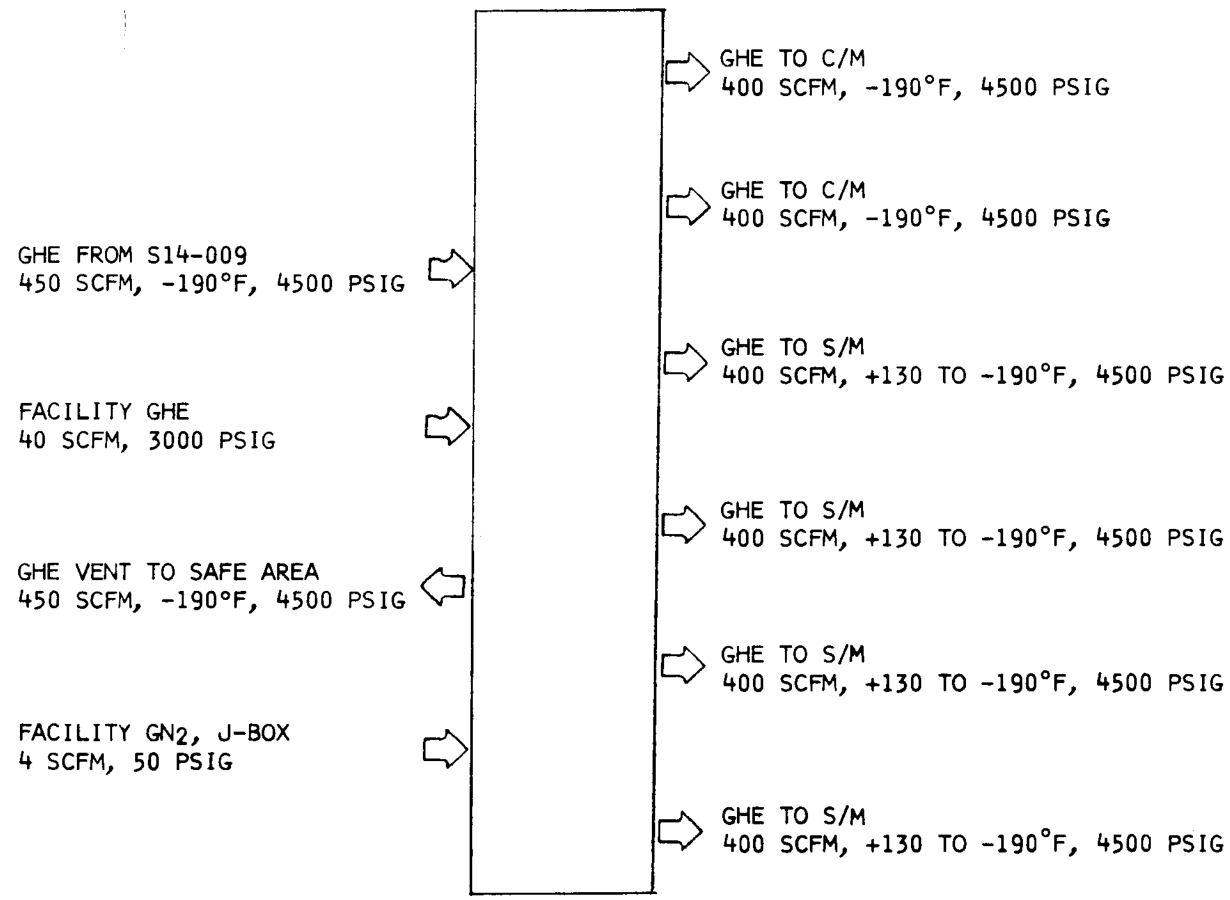
SPS Helium and GN_2 Valve Box

These units, with associated fluid lines and flex hoses, interface with the Spacecraft systems; provide either remote or local control capability; and are the end items for the specialized fluid loading systems. The fluid system capabilities for each item are shown in the following diagrams. These units have application in the Voyager Program due to the general similarity of the gaseous requirements. Use of these items in the Voyager Program would depend upon further investigation into the areas of (1) compatibility and modifications for the vehicle interface to the equipment; (2) operational usage and procedural control methods acceptable to the Voyager Program; and (3) specific gaseous servicing times and deliverable fluid conditions for the spacecraft requirements.

SPS HELIUM & GN_2 VALVE BOX
8G14-840099



RCS HELIUM VALVE BOX
8G14-840023



HELIUM VALVE/CONTROL BOXES

EQUIPMENT SUMMARY DATA SHEET
Sheet 17 of 82

APOLLO FUNCTION SUMMARY

This unit supplies GN_2 to the Apollo Spacecraft Service Propulsion System (SPS) propellant tanks during storage and transportation in order to ensure structural integrity and protection from contamination.

APOLLO REQUIREMENTS

GN_2 is provided at 300 to 3,000 psi from GN_2 bottles or from a facility supply. Outlet GN_2 pressure is regulated by the unit at 10 $\pm$ 5 psig and the system relieves at 35 $\pm$ 5 psig. Nitrogen per MIL-P-27401 is used. Materials are also compatible with N_2O_4 and aerozine 50.

PHYSICAL SIZE

Length - 33 inches	Height - 62 inches
Width - 26 inches	Weight - 150 pounds without bottles

SUPPLY REQUIREMENTS

Two bottles GN_2, or connection to facility GN_2.

VOYAGER EVALUATION

Portable pressurization equipment will be required for the Voyager during inactive periods of handling and testing. This unit will meet those requirements but cleanliness of the propellant systems may require additional filtering systems for spacecraft protection.

S14-099 PROTECTIVE PRESSURIZATION UNIT ASSEMBLY
PART NO. G17-848540 (NAA)

EQUIPMENT SUMMARY DATA SHEET
Sheet 18 of 82

3.0 SPACECRAFT CHECKOUT EQUIPMENT

3.1 Spacecraft Test

Helium Leak Tester Mass Spectrometer, S14-003 (NAA)

Propulsion System Fluid Checkout Unit Console Assembly, C14-075-301 (NAA)

Adapter Unit - PUGS SPS, C14-352 (NAA)

Descent Engine Simulator Live Propellant, LDW-430-6150 (GAEC)

Thrust Chamber Assembly Alignment Equipment, C14-408-0001 (NAA)

Pyrotechnic Initiators Substitute Unit, A14-003 (NAA)

Pyrotechnic Initiators Substitute Unit, A14-139 (NAA)

Optical Alignment Set, A14-028 (NAA)

Electronic Weighing Kit, H14-041 (NAA)

Box Level, A14-047 (NAA)

AOT Optical Target Set, LDW 420-13371 (GAEC)

Descent Engine Plug Assembly, 420-63420 (GAEC)

Descent Engine Leak Test Set, 420-62366 (GAEC)

Mobile Optical Alignment Equipment, 420-13360 (GAEC)

3.2 Auxiliary Test

Engine Decontamination Unit, S14-070 (NAA), LSC 430-94070 (GAEC)

RCS Freon Flush Cart, LDW 430-6860 (GAEC)

RCS Engine and Purge Unit - Oxidizer, LDW 430-2490 (GAEC)

Flowmeter Cart, 30-60 scfm, 430-6420 (GAEC)

Calibration Unit - Pressure, 6000 psig, C14-426 (NAA)

Flow Rate Calibration Set, C14-427 (NAA)

Temperature Calibration Set, C14-428 (NAA)

Ground Cooling Cart, A14-011 (NAA)

Ground Air Circulating Unit, A14-036 (NAA)

Vacuum Cleaner, A14-035-0002 (NAA)

Temperature Controlled Battery Storage Rack, 420-83280 (GAEC)

3.3 Test Fixtures and Stands

C/M Optical Alignment Support Equipment, A14-135 (NAA)

Base Support Stand, H14-031 (NAA)

Command Module Maintenance Stand, H14-035 (NAA)

Access Stand for External S/C, H14-109-101 (NAA)

Spacecraft Integrated Systems Workstands, H14-124 (NAA)

S/M and S/C Adapter Weight and Balance Fixture, H14-9059 (NAA)

Weight and Balance Jack Set, H14-9015-101 (NAA)

Weight and Balance Fixture, LDW 420-13460 (GAEC)

Descent Engine Support Fixture 420-6043 (GAEC)

Descent Stage Propellant Tanks Installation Fixture, LSC 420-63150 (GAEC)

LEM Turnover and Handling Fixutre, LDW 420-13110 (GAEC)

Three Axis Positioner Work Platform, 420-13220 (GAEC)

LEM Integrated Workstand, 420-13390 (GAEC)

Cleaning Positioner, LDW 420-13750 (GAEC)

Polarity Checker, LSC-420-93089 (GAEC), G14-818100 (NAA)

Polarity Checker Work Platform, 420-31040 (GAEC)

Optical Alignment Fixture, LDW 420-13360 (GAEC)

APOLLO FUNCTION SUMMARY

Provides rapid means of detecting leakage of the spacecraft ECS, electrical power systems, fluid systems including reacting circuits, SPS, and the RCS. The mass spectrometer is used in leak test checkout of the spacecraft and subassemblies during final assembly in the manufacturing area and in the MSOB prelaunch checkout at KSC.

APOLLO REQUIREMENTS

A. Unit must have a basic sensitivity of 5 X 10^{-10} standard cc/sec and be capable of locating a leak magnitude of 1 X 10^{-6} standard cc/sec.

B. Insensitive to gases other than helium.

C. Unit weight 656 pounds. Cabinet size - 37 length X 32 width x 48 height.

D. Inputs: 115 volts, 60 cycle single phase.

VOYAGER REQUIREMENTS

Using helium as a test medium, the Voyager leak test unit must be capable of detecting 20 ppm.

S14-003 HELIUM LEAK TESTER MASS SPECTROMETER
PART NO. G14-848001 (NAA)

EQUIPMENT SUMMARY DATA SHEET
Sheet 19 of 82

APOLLO FUNCTION SUMMARY

This unit filters and regulates gaseous helium, freon, and nitrogen supplied to the F-2 test fixture during operational and leak checking of the SPS and RCS at Downey, California. A unit is also located on the IVA platform of the MSS at KSC for operational support of the prelaunch testing. It interfaces with the MSS gas distribution system.

APOLLO REQUIREMENTS

A. Filter helium to 5 micron nominal - 15 micron absolute.

B. Shall regulate, control, and indicate pressures in the propellant pressure systems of the RCS and SPS in the 0 to 2,000 psig pressure range to a 2-percent accuracy above 500 psig and 5 percent accuracy below 500 psig.

C. Readouts shall be 0.2 percent accurate above 500 psig.

D. Shall accept 0 to 6,000 psig facility helium.

SUPPLY REQUIREMENT

0 to 6,000 psig helium.

PHYSICAL SIZE

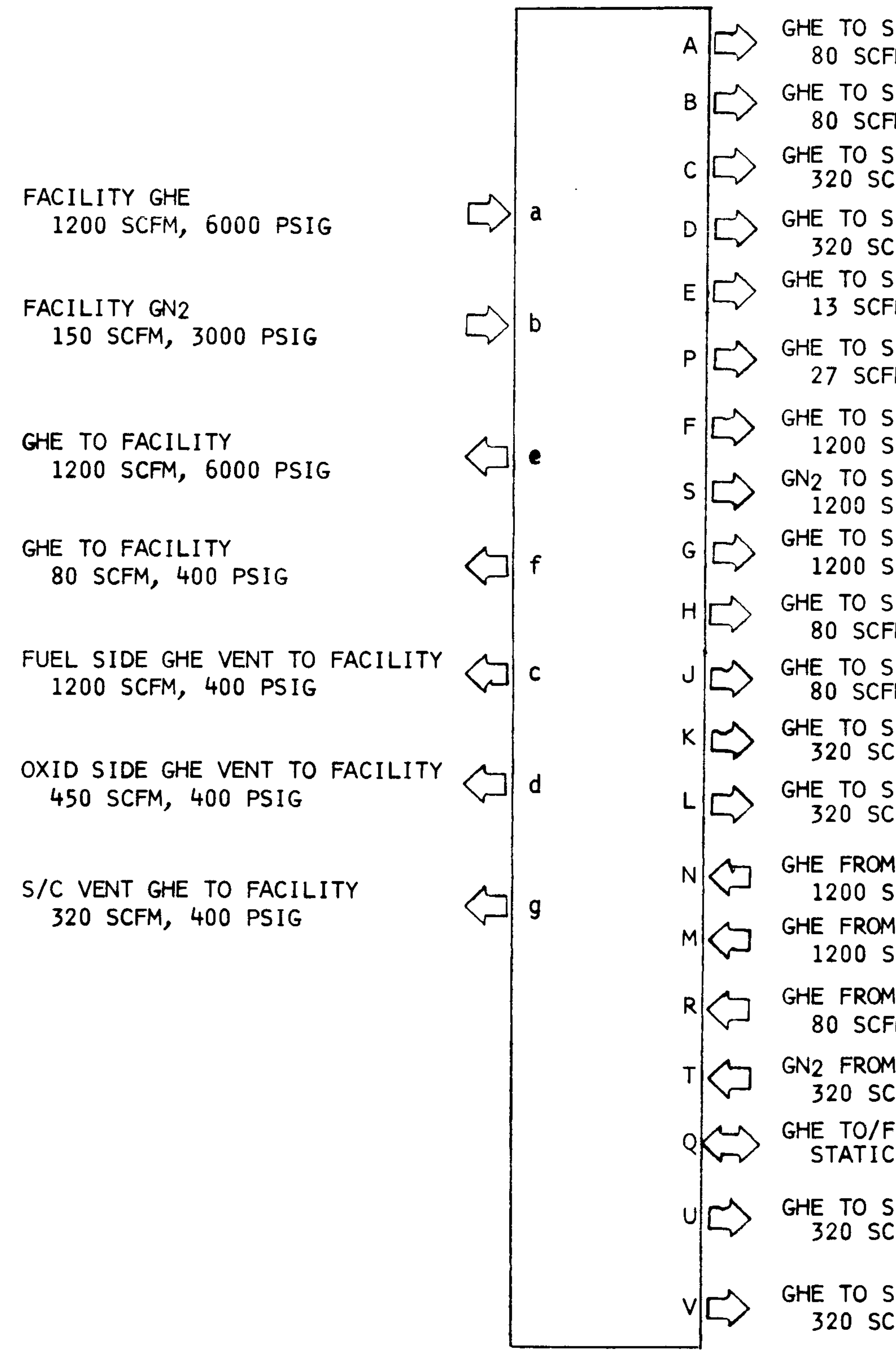
Length - 62 inches

Width - 30 inches.

Height - 62 inches

Weight - 1,000 pounds.

C14-075-201 PROPULSION SYSTEM CHECKOUT



FOLDOUT FRAME

D-26A

FOLDOUT FRAME

D-26B

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VOYAGER EVALUATION

The unit is applicable for use in the Voyager Program at the Valley Forge facility and KSC with the proper hose assemblies for interfacing with the Voyager fluid systems.

C14-075-301	PROPULSION SYSTEM FLUID CHECKOUT UNIT CONSOLE ASSEMBLY PART NO. G17-854300-301 (NAA)

EQUIPMENT SUMMARY DATA SHEET
Sheet 20 of 82

FOLDOUT FRAME

APOLLO FUNCTION SUMMARY

This unit provides remote control for ACE to adjust, calibrate, and checkout the SPS Propellant Utilization Gauging System (PUGS). It is used at the launch pad, the Hypergolic Test Facility and the MSOB checkout area.

APOLLO REQUIREMENTS

A. Provide ACE with 5 percent propellant level during dry state checkout operation of PUGS.

B. Monitor propellant level during wet state checkout operation of PUGS.

C. Hazardous purge protection.

SUPPLY REQUIREMENT

Interface with ACE, SPS fuel and oxidizer tanks, facility power and GN_2.

PHYSICAL SIZE

Weight - 80 pounds.

VOYAGER EVALUATION

The configuration of the Voyager Spacecraft Propellant Measuring System has not been defined. This unit cannot be evaluated at present because of unknown vehicle interface requirements for ACE and the propellant systems.

C14-352 ADAPTER UNIT - CHECKOUT PUGS SPS
PART NO. G17-854620 (NAA)

EQUIPMENT SUMMARY DATA SHEET
Sheet 21 of 82

LEM FUNCTION SUMMARY

This unit is used to functionally check out the feed system without the use of an actual engine and to size orifices and balance the propellant feed system line runs to meet individual engine interface requirements. It is used primarily in the Bethpage, Long Island manufacturing facility.

LEM REQUIREMENTS

A. Inlet pressure ranges 0 to 220 psi.

B. Flow adjustment 10 to 110 gpm flowmeters to $\pm$ 0.25 percent accuracy.

C. Shutoff valve reaction time of 100 to 1,000 milliseconds.

D. Compatible with live and substitute propellants.

SUPPLY REQUIREMENTS

A. 115 vac, one phase, 60 cycle.

B. 28 vdc.

C. GN_2 valve actuation supply.

PHYSICAL SIZE

Length - 40 inches.

Width - 28 inches.

Height - 60 inches.

Weight - 600 pounds.

VOYAGER EVALUATION

This has direct application in the Voyager Program if the (STL) descent engine is part of the "bus" configuration. Flowmeter accuracy may require improvement and the interface connecting points to the vehicle propellant feed lines may necessitate modifications.

DESCENT ENGINE SIMULATOR LIVE PROPELLANT PART NO. LDW 430-6150 (GAEC)

EQUIPMENT SUMMARY DATA SHEET
Sheet 22 of 82

APOLLO FUNCTION SUMMARY

This unit will be used during engine assembly and at bench maintenance areas where thrust chambers, gimbal bearings, and propellant lines may be replaced. The unit will be used as GSE for the SPS engine in the Spacecraft Checkout Area of the MSOB.

APOLLO REQUIREMENTS

A. Combustion chamber alignment to the gimbal ring and propellant line to within 0.010 in X, Y, and Z axes.

B. Unit is detachable and portable.

VOYAGER EVALUATION

Engine alignment will be required on the Voyager Program. This unit could be considered dependent upon the final selection of the Voyager engine configuration. However, additional jigs or fixtures may be required for application to another engine assembly.

C14-408-0001	THRUST CHAMBER ASSEMBLY ALIGNMENT EQUIPMENT, BENCH MAINTENANCE PART NO. MC 901-0147 (NAA)

EQUIPMENT SUMMARY DATA SHEET
Sheet 23 of 82

APOLLO FUNCTION SUMMARY

This unit determines that electrical inputs to hot and explosive bridge wire squibs are of required magnitude and duration to ensure firing of these devices during a launch. Inputs originate in the sequencing circuits of the C/M. The unit is used primarily in the Spacecraft checkout area of the MSOB.

APOLLO REQUIREMENTS

A. Must indicate a GO signal upon receipt of a 5 $\pm$ 0.1 amp signal for 18 millisecond maximum.

B. Records a transient voltage level greater than 2.5 $\pm$ 1 vdc.

C. Provides substitute units for airborne systems, with a central control console.

SUPPLY REQUIREMENTS

115 $\pm$ 10 vac, 60 cps.

PHYSICAL SIZE

Weight - 250 pounds (control console).

VOYAGER EVALUATION

The checkout of the Voyager Spacecraft will require a similar unit as described above. However, the sub-modules would not be adaptable to the Voyager system and the squib system acceptance levels may be different from the capabilities and thus require modification.

A14-003	PYROTECHNIC INITIATORS SUBSTITUE UNIT PART NO. G16-820500 (NAA)

EQUIPMENT SUMMARY DATA SHEET
Sheet 24 of 82

APOLLO FUNCTION SUMMARY

This unit provides non-destructive substitutes for the pyrotechnic initiators. The initiators substitute unit shall provide instrumentation feedback during test. It is used primarily in the Spacecraft checkout area of the MSOB.

APOLLO REQUIREMENTS

A. Shall accept input signals 0 to 37 vdc from the Spacecraft pyro system.

B. Transient signals will be indicated for 0.15 minus 5.0 vdc or less than minus 0.15 vdc.

C. "GO" signals when signals are 5.0 plus 0.05 minus zero or greater for 10 milliseconds.

D. Simulate burnout characteristics of initiator by reducing firing current to an open circuit within 100 milliseconds.

SUPPLY REQUIREMENTS

28 vdc ± 4 volts.

PHYSICAL SIZE

Length - 12 inches.
Width - 10 inches.
Height - 7 inches.
Weight - 20 pounds.

VOYAGER EVALUATION

This unit should be considered for use in the Voyager Program but must be reviewed for system compatibility after Voyager ordnance requirements are defined. Items to be reviewed for compatibility would be unit resistance, reset capability, accidental firing simulation, misfire simulation and stray voltage indication.

A14-139	PYROTECHNIC INITIATORS SUBSTITUTE UNIT PART NO. G16-820402 (NAA)

APOLLO FUNCTION SUMMARY

This unit shall provide precise vertical planes for alignment applications on the C/M and the LES in the Spacecraft checkout area of the MSOB.

APOLLO REQUIREMENTS

A. Telescopic.

B. Auto-collimation and auto-reflection.

C. Optical micrometer.

D. Leveling capability.

E. 180-degree minimum sweep.

F. Establish vertical planes within 6 second of arc and level planes within 2 second of arc.

VOYAGER EVALUATION

This unit would have applications in the weight and C.G. determinations of the Voyager Spacecraft during checkout. Adapters for specific equipment mounting will be necessary. Maximum Voyager tolerance for use of this unit is $\pm$ 15 arc seconds during Midcourse Engine Alignment. Other applications would be antenna and capsule interface alignments.

A14-028	OPTICAL ALIGNMENT SET PART NO. G17-880130 (NAA)

EQUIPMENT SUMMARY DATA SHEET

Sheet 26 of 82

APOLLO FUNCTION SUMMARY

This unit measures spacecraft weight and permits weight and center of gravity calculations for C/M, adapter, and the LES. It is a portable unit and is used in the Spacecraft checkout area of the MSOB.

APOLLO REQUIREMENTS

A. Capacity to 30,000 pounds.

B. Can be used with dc digital indicator, A14-154.

C. Consists of electronic force indicator and three load cells which are installed on various weighing fixtures.

D. Allowable system error within $\pm$ 0.1 percent of applied load or one digital increment, whichever is greater.

E. Portable.

PHYSICAL SIZE

Weight - 77 pounds.

SUPPLY REQUIREMENTS

120 vac, 60 cps, single phase.

VOYAGER EVALUATION

Unit can be used without modifications on the Voyager Spacecraft. Adapters will be necessary to mount the load cells on any newly designed Voyager weighing fixture.

H14-041	ELECTRONIC WEIGHING KIT PART NO. G14-810095 (NAA)

EQUIPMENT SUMMARY DATA SHEET
Sheet 27 of 82

APOLLO FUNCTION SUMMARY

This unit provides a capability for establishing vertical and horizontal reference planes for alignment purposes on the LES and Spacecraft modules. It is used with alignment fixtures in the Spacecraft checkout area of the MSOB.

APOLLO REQUIREMENTS

Indicate a true horizontal and vertical plane within 2 seconds of arc.

PHYSICAL SIZE

Length - 4.80 inches.
Width - 1.62 inches.
Height - 3.5 inches.
Weight - 2 pounds.

VOYAGER EVALUATION

This unit can be used for the Voyager Program without modification. However, proper interface design with the special test jigs would be necessary if this unit is to be used in such items as alignment fixtures and the Spacecraft assembly fixture.

A14-047	BOX LEVEL PART NO. G15-820045 (NAA)

EQUIPMENT SUMMARY DATA SHEET
Sheet 28 of 82

LEM FUNCTION SUMMARY

This unit provides an earth reference coordinate system to verify the over-all alignment between the AOT and IMU stable platform as well as optical alignment of the G&N subsystems during prelaunch checkout test in the KSC MSO Building.

LEM REQUIREMENTS

Three theodolites mounted along the line of sight of the AOT at the three detent positions.

PHYSICAL SIZE

Length - 30 inches.
Width - 12 inches.
Height - 28 inches.
Weight - 80 pounds.

VOYAGER EVALUATION
This unit is applicable in the Voyager Program but would require modification to correspond with Voyager alignment requirements. New fixture equipment would probably be required to accommodate the larger Voyager vehicle diameter.

AOT OPTICAL TARGET SET
PART NO. LDW 420-13371

EQUIPMENT SUMMARY DATA SHEET
Sheet 29 of 82

LEM FUNCTION SUMMARY

This unit is used to verify the pressure integrity of the descent engine and related integral piping by sealing the throat of the engine while on a maintenance stand or in the vehicle at the Spacecraft checkout facility of the MSO Building at KSC.

LEM REQUIREMENTS

A. Seal the engine throat area.

B. Provide ingress capability for helium mass spectrometer for leak detection, vacuum roughing pump, and pressure transducers.

C. Self-supporting with a relief device.

SUPPLY REQUIREMENTS

130 psig GN_2.

PHYSICAL SIZE

Length - 70 inches.

VOYAGER EVALUATION

This unit will have direct applicability in the Voyager Program if the STL descent engine is used for the Voyager "bus."

DESCENT ENGINE PLUG ASSEMBLY
PART NO. 420-63420 (GAEC)

EQUIPMENT SUMMARY DATA SHEET
Sheet 30 of 82

LEM FUNCTION SUMMARY

This unit provides a means of leak checking the thrust chamber to nozzle extension flange joint after engine build-up at the Spacecraft checkout facility of the MSO Building at KSC.

LEM REQUIREMENTS

A. Pressure check the flange joint while the engine is in the propulsion system or dolly mounted.

B. Self-sealing.

SUPPLY REQUIREMENTS

A. Shop air 150 psig.

B. GN_2 150 psig.

PHYSICAL SIZE

Length - 72 inches.

VOYAGER EVALUATION

This unit should be used in the Voyager Program if the STL descent engine is selected as part of the Voyager "bus" configuration.

DESCENT ENGINE LEAK TEST SET PART NO. 420-62366 (GAEC)

EQUIPMENT SUMMARY DATA SHEET
Sheet 31 of 82

LEM FUNCTION SUMMARY

This unit provides the capability to optically verify the mechanical angular alignment of certain components with respect to the vehicle reference frame. This is required for the rendezvous and landing radar antenna, docking and landing reticles, RCS engine assembly thruster axis, and initial descent engine gimbal bias angle. This alignment is accomplished in the Spacecraft checkout area of the KSC MSO Building.

LEM REQUIREMENTS

A. One order of accuracy greater than the alloted error of the mechanical angular measurement.

B. Mobile access platforms containing a mobile vertical tooling bar with leveler, theodolite, and electronic level.

PHYSICAL SIZE

Length - 12 feet.
Width - 5 feet.
Height - 25 feet.

VOYAGER EVALUATION

This unit can be used without physical modifications on the Voyager Program. However, the Voyager alignment accuracy requirements have not yet been defined for all systems requiring alignment.

MOBILE OPTICAL ALIGNMENT EQUIPMENT
PART NO. 420-13360 (GAEC)

EQUIPMENT SUMMARY DATA SHEET
Sheet 32 of 82

APOLLO FUNCTION SUMMARY

The unit provides decontamination of the SPS engine system prior to the removal of the SPS engine or engine components. The unit is portable and can be used in the Hypergolic Propellant Facilities or the Propulsion Test Facilities.

APOLLO REQUIREMENTS

A. Pump a flush fluid (methyl alcohol).
B. Pump a flush fluid (freon FM) at 70 ± 2 gpm against a 90-foot freon MF head.
C. Monitor flowrates (liquid) to ± 10 percent accuracy.
D. Heat gaseous nitrogen to 160 ± 10 degrees F.
E. Control GN_2 at 400 scfm flow rate and pressures to 184 psig.
F. Gas and liquid filters to remove 15 micron absolute and 5 micron nominal.
G. Wheel mounted with tow bar.

VOYAGER REQUIREMENTS

The scrub of a Voyager launch after the completion of propellant servicing would require a decontamination of the system prior to any demating or disassembly from the launch vehicle.

SUPPLY REQUIREMENTS

A. GN_2 at 1,800 to 3,000 psig between 15 degrees F. and 105 degrees F.
B. Liquid methyl alcohol at 4 feet head.
C. Liquid freon FM at 4 feet head.
D. 480 Y/277 volts, 3 phase, 60 cps, 3 wire.

PHYSICAL SIZE

Length - 72 inches.
Width - 50 inches.
Height - 64 inches.
Weight - 3,400 pounds.

FOLDOUT FRAME

D-39A

VOYAGER EVALUATION

The Engine Decontamination Unit has application for use on Voyager depending upon the flush fluids and flush sequence used for Voyager. If methyl alcohol, freon MF and hot GN_2 are used, this unit would be acceptable for Voyager without modification.

S-14-070	ENGINE DECONTAMINATION UNIT PART NO. G17-848180 (NAA) LSC 430-94070 (GAEC)

EQUIPMENT SUMMARY DATA SHEET
Sheet 33 of 82

FOLDOUT FRAME

APOLLO FUNCTION SUMMARY

The unit provides either freon or deionized water for flushing of the LEM RCS manifolds. It is portable and is used on the MSS platform III to service the Spacecraft.

APOLLO REQUIREMENTS

A. Store 85 gallons of freon or deionized water.
B. Flow and pressure control.
C. Measure tank level and flow rate (0 to 10 gpm).
D. Recirculation.
E. 10 micron absolute filtration.
F. Condition water to 120 degrees F. maximum.

PHYSICAL SIZE

Length - 5 feet.
Width - 3 feet.
Height - 5 feet.
Weight - 1,200 pounds.

SUPPLY REQUIREMENTS

208-volt, 60 cps, 3 phase.

VOYAGER EVALUATION

The decontamination requirements of the Voyager AC Systems are at present undefined. However, if "propellant-type" contaminates are to be removed, consideration should be given to the usage of the above unit, with or without modifications other than for interface connections.

RCS FREON (OR DEIONIZED WATER) FLUSH CARTS
PART NO. LDW 430-6860 (GAEC)

EQUIPMENT SUMMARY DATA SHEET
Sheet 34 of 82

APOLLO FUNCTION SUMMARY

Provides a flush, purge and evacuation of the LEM RCS (ascent and descent) engine oxidizer systems. The units are portable and are used on level III platform of the MSS and in the Spacecraft checkout area of the MSOB.

The fuel unit serves the same function for fuel as the LDW 430-2490 oxidizer unit. The major significant difference would be the utilization of two flush fluids; methyl alcohol and freon MF, for fuel systems flushing. Each fluid storage capacity is 50 gallons within the unit.

The considerations for usage of this unit in the Voyager Program would be the same as for the oxidizer unit.

APOLLO REQUIREMENTS

A. Provide 100 gallon freon MF storage capacity.

B. Provide GN_2 at 0 to 225 psig and 70 degrees F. to 160 degrees F.

C. Provide water jet aspirator capable of 23 to 27 inches H_g vacuum on a closed suction line.

D. Mobile (wheel mounted).

SUPPLY REQUIREMENTS

Unknown quantities at this time.

PHYSICAL SIZE

Length - 85 inches.
Width - 48 inches.
Height - 69 inches.
Weight - 1,500 pounds.

VOYAGER EVALUATION

The scrub of a Voyager launch after the completion of propellant tanking would require a decontamination of any "propellant-type" AC system.

The unit should be considered for Voyager applications dependent upon the type of AC propellants used on Voyager. Flushing medians would have to be reviewed for Voyager systems compatibility also.

RCS ENGINE FLUSH AND PURGE UNIT - OXIDIZER PART NO. LDW 430-2490 (GAEC)

EQUIPMENT SUMMARY DATA SHEET
Sheet 35 of 82

LEM FUNCTION SUMMARY

This unit provides for glass tube flowmeters to measure gas flow of 30 to 60 scfm during functional tests of prototypes and flight weight components in the support labs for Spacecraft checkout in the KSC MSO Building.

LEM REQUIREMENTS

A. Measure helium flows from 2.97 to 567.0 scfm with three glass tube flowmeters.

B. Tube accuracy of 1.0 percent of scale reading.

C. Double-scaled tube for direct readings in scfm of helium and nitrogen.

PHYSICAL SIZE

Length - 34 inches.

Width - 38 inches.

Height - 81 inches.

Weight - 250 pounds.

VOYAGER EVALUATION

This unit fan be used for similar application in the Voyager Program but the measurement accuracies should be checked against Voyager requirements.

FLOWMETER CART 30 to 60 SCFM
PART NO. 430-6420 (GAEC)

EQUIPMENT SUMMARY DATA SHEET
Sheet 36 of 82

APOLLO FUNCTION SUMMARY

This unit calibrates Spacecraft and GSE pressure gages and transducers. It is portable and can be used in all prelaunch checkout of the Spacecraft at the MSOB and the VAB.

APOLLO REQUIREMENTS

A. Calibration range of 0 to 6,000 psig with 0.17 percent of full scale accuracy.

B. Mobile on casters.

SUPPLY REQUIREMENTS

A. 120/208 vac, 3 phase, 60 cps.

B. 7,000 psig GN_2 at 350 scfm.

PHYSICAL SIZE

Length - 72 inches.

Width - 34 inches.

Height - 62 inches.

Weight - 3,400 pounds (two pieces).

VOYAGER EVALUATION

This unit can be used on the Voyager Program with only minor modifications to interface connections to Voyager equipment or OSE.

C14-426	CALIBRATION UNIT - PRESSURE 6,000 PSIG PART NO. G16-854100 (NAA)

EQUIPMENT SUMMARY DATA SHEET
Sheet 37 of 82

APOLLO FUNCTION SUMMARY

This unit calibrates flow rate instrumentation in GSE for gas and liquid systems. It is a mobile unit primarily used in the Spacecraft checkout areas of the VAB and MSOB.

APOLLO REQUIREMENTS

A. Simulate liquid flow from 0.2 to 70 sgpm by centrifugal pump and metering valves with digital readouts from turbine type flow meters.

B. Simulate gas flow with GN_2 at 70 degrees F. at ranges from 0.5 scfm to 6000 scfm with readout on digital displays from turbine type flow meters or differential pressure flow meters in standard cubic centimeters per minute.

C. Liquid flow accuracy of 0.75 percent of indicated; gaseous flow accuracy of 1.5 percent of indicated; gage accuracy of $\pm$ 0.1 percent of full scale.

D. Particle count of test fluid:

Liquid - from 1 to 2 X 10^4 particles/100 ml.

Gaseous - from 1 to 3 X 10^6 particles/scf.

E. Unit internal hazardous purge.

F. Deionized water pump to deliver 70 gpm at 165 feet head with a maximum npsh of 8 feet.

SUPPLY REQUIREMENTS

A. Propellant systems interface.

B. ECS interface.

C. GH_e and GN_2.

D. Facility power and ground.

PHYSICAL SIZE

Length - 72 inches.

Width - 35 inches.

Height - 65 inches.

Weight - 2,200 pounds.

VOYAGER EVALUATION

This unit should be considered as a requirement for the Voyager Program. Propellant loading accuracies desired for Voyager, however, indicate that the calibration accuracies of this unit would have to be improved for use in the Voyager systems.

C14-427 FLOW RATE CALIBRATION SET
PART NO. G16-854580 (NAA)

EQUIPMENT SUMMARY DATA SHEET
Sheet 38 of 82

APOLLO FUNCTION SUMMARY

This unit provides controlled temperatures used in calibrating Spacecraft and GSE temperature sensing instrumentation. It is a mobile unit used primarily in Spacecraft checkout areas of the MSOB and the VAB.

APOLLO REQUIREMENTS

A. Control the unit internal temperature between minus 90 degrees F. to plus 300 degrees F.

B. Monitor output voltage of test unit.

C. Provide 280 vdc ± 4 excitation voltage to the test unit.

SUPPLY REQUIREMENTS

115 ± 10 vac, 60 cps, 3 phase.

PHYSICAL SIZE

Weight - 1,500 pounds.

VOYAGER EVALUATION

This unit is applicable for use in the Voyager Program. The calibration accuracy, at present unknown, should be evaluated against Voyager requirements. Voyager pressurant loading temperature indications shall have an accuracy of ± 2.5 degrees.

C14-428	TEMPERATURE CALIBRATION UNIT PART NO. G16-854300 (NAA)

EQUIPMENT SUMMARY DATA SHEET
Sheet 39 of 82

APOLLO FUNCTION SUMMARY

The unit is portable and provides conditioned air to the interior of the C/M for control of interior temperature, humidity, and cleanliness during preflight periods of test and maintenance in the MSOB, the VAB, or other test facilities.

APOLLO REQUIREMENTS

A. Capable of receiving 105 degrees F. 65 percent relative humidity ambient air and delivering 23 pounds per minute at 75 degrees F. and less than 50 percent relative humidity at a minimum pressure of 3 inches of water.

B. Capable of heating air from 15 degrees F., 100 percent relative humidity to 95 degrees F. at the above capacity.

SUPPLY REQUIREMENTS

480 vac, 3 phase, 4 wire, 60 cycle.

PHYSICAL SIZE

Length - 58 inches.
Width - 40 inches.
Height - 72 inches.
Weight - 3,000 pounds.

VOYAGER EVALUATION

There is no specific application for this unit on the Voyager Program but cleanliness controls will be similar. Interconnecting duct work would probably require modifications if this unit were used in support of Voyager Spacecraft checkout.

A14-011	GROUND COOLING CART PART NO. ME 362-0002-0001 (NAA)

EQUIPMENT SUMMARY DATA SHEET
Sheet 40 of 82

APOLLO FUNCTION SUMMARY

This is a mobile unit to supply fresh ambient air to the C/M during testing, maintenance, and checkout in all test areas.

APOLLO REQUIREMENTS

Capable of delivering 800 cfm of ambient air.

SUPPLY REQUIREMENTS

Length - 24.5 inches.

Width - 12.5 inches.

Height - 21.5 inches.

Weight - 60 pounds.

VOYAGER EVALUATION

This unit should be considered for use in the Voyager Program in its present configuration as a piece of general use equipment. Its application would be to supply air to enclosed or minimum clearance areas in which men are working.

A14-036 GROUND AIR CIRCULATING UNIT
PART NO. G16-824150-301 (NAA)

EQUIPMENT SUMMARY DATA SHEET
Sheet 41 of 82

APOLLO FUNCTION SUMMARY

This is a portable unit used to clean the interior of the C/M in all test and checkout areas.

APOLLO REQUIREMENTS

A. Incorporate a 30-foot extension hose.

B. Lift capability of 47 inches of water.

C. Six-gallon refuse container.

D. Portable.

SUPPLY REQUIREMENTS

480 volts, 3 phase, 60 cps, 4 wire.

PHYSICAL SIZE

Length - 66 inches.

Width - 26 inches.

Height - 53 inches.

Weight - 400 pounds.

VOYAGER EVALUATION

This unit could be used in the Voyager Program without modifications as a general purpose cleaning unit during Spacecraft checkout.

A14-035-0002 VACUUM CLEANER
PART NO. ME 901-0064-0002 (NAA)

EQUIPMENT SUMMARY DATA SHEET
Sheet 42 of 82

LEM FUNCTION SUMMARY

This unit provides for cold storage (35 degrees F. to 45 degrees F.) for two sets of flight batteries after the batteries have been activated at the Spacecraft checkout facility in the KSC MSO Building.

LEM REQUIREMENTS

A. Maintain 35 degrees to 45 degrees F.

B. Capable of storing two sets of batteries and ERA handling fixture.

C. Temperature recording device.

SUPPLY REQUIREMENTS

208 vac, 3 phase, 60 cycle.

PHYSICAL SIZE

Length - 15 feet.

Width - 15 feet.

Height - 8 feet.

VOYAGER EVALUATION

This unit should be considered for use in the Voyager Program if the battery requirements dictate such a need.

TEMPERATURE CONTROLLED BATTERY STORAGE RACK PART NO. 420-83280 (GAEC)

EQUIPMENT SUMMARY DATA SHEET
Sheet 43 of 82

APOLLO FUNCTION SUMMARY

This unit supports and positions guidance and navigation optical alignment equipment, and is used in support of the optical alignment kit and other fixtures in the Spacecraft checkout area of the MSOB.

APOLLO REQUIREMENTS

A. Support the command module.

B. Capability to level to C/M optical alignment with respect to gravity, within ± 2 arc seconds and horizontally with respect to the established azimuth within ± 8 arc seconds.

PHYSICAL SIZE

Length - 120 inches.

Width - 68 inches.

Height - 72 inches.

Weight - 3,035 pounds.

VOYAGER EVALUATION

The Voyager Program will require similar equipment but at present these requirements have not been defined. It is also doubtful if this unit could be modified to facilitate the handling of the larger Voyager Spacecraft.

A14-135	C/M OPTICAL ALIGNMENT SUPPORT UNIT PART NO. G16-828160

EQUIPMENT SUMMARY DATA SHEET
Sheet 44 of 82

APOLLO FUNCTION SUMMARY

The stand provides access, as well as personnel and equipment support for servicing and checkout of the service module within the clean room area of the Spacecraft Checkout Building (MSOB).

APOLLO REQUIREMENTS

A. Fixed floor live loading support of 60 psf plus concentrated dead load of 1,000 pounds/6" X 6".

B. Movable floor live loading support of 60 psf plus a concentrated dead load of 300 pounds/6" X 6".

C. 500 pounds integral hoist.

D. Interface with S14-014 and -082 fluid distribution systems.

E. Two working levels.

SUPPLY REQUIREMENTS

A. 120/208 vac, 60 cycle, 3 phase.

B. Compressed air.

C. Water glycol and GN_2.

PHYSICAL SIZE

Length - 333 inches.

Width - 284 inches.

Height - 330 inches.

Weight - 40,400 pounds.

VOYAGER EVALUATION

Load capacities of the stand would support the Voyager Spacecraft. However, platform level working heights appear to be too high as well as the opening diameters in the platforms being too small to accept a Voyager Spacecraft. Modifications of the stand would render the item as unserviceable for the Apollo and would require complete structural design review to evaluate load capacities after alterations. Therefore, this item should not be used on Voyager.

H14-031	BASE SUPPORT STAND PART NO. G17-818093 (NAA)

APOLLO FUNCTION SUMMARY

This stand provides access for personnel and equipment for servicing and checkout of the command module within a clean area of the Spacecraft Checkout Building (MSOB).

APOLLO REQUIREMENTS

A. Two major levels of platforms.

B. Capable of supporting 11,000 pounds maximum static load.

C. Support 60 psf live load and 300 pounds/6" X 6" dead load on stairs and platforms.

PHYSICAL SIZE

Length - 246 inches.

Width - 246 inches.

Height - 150 inches.

Weight - 23,000 pounds.

VOYAGER EVALUATION

The checkout and maintenance stand concept is acceptable to the Voyager Program but the H14-035 unit would require extensive structural modifications and platform adjustments in order to accept the Voyager Spacecraft. These modifications would render the unit not usable for Apollo. Therefore, the stand should not be used.

H14-035	COMMAND MODULE MAINTENANCE STAND PART NO. G16-818003 (NAA)

EQUIPMENT SUMMARY DATA SHEET
Sheet 46 of 82

APOLLO FUNCTION SUMMARY

This unit provides a 360-degree working access around the forward attachment interface of the S/M while it is resting on a support base and/or dolly in the Spacecraft checkout area of the MSOB.

APOLLO REQUIREMENTS

A. Interfaces with the S14-036 FDS unit and A14-011 ground cooling cart.

B. Twenty feet above ground level work platform.

C. Load capacity of 60 psf plus a dead load of 300 pounds/6 X 6 inch area.

PHYSICAL SIZE

Work platform area is 19.5 feet square.

Weight is 6,100 pounds.

SUPPLY REQUIREMENTS

120/208 vac, 3 phase, 60 cps, 4 wire.

VOYAGER EVALUATION

This unit must be reviewed more extensively for Voyager usage due to a Spacecraft size differential. Modifications would be required, in any event, to use this on the Voyager Program.

H14-109-101	ACCESS STAND FOR EXTERNAL S/C PART NO. G17-810070-101 (NAA)

EQUIPMENT SUMMARY DATA SHEET
Sheet 47 of 82

APOLLO FUNCTION SUMMARY

This unit provides four platform working levels which encompass an assembled Spacecraft in the Spacecraft checkout area of the MSOB.

APOLLO REQUIREMENTS

A. Lightweight aluminum alloy construction capable of rapid disassembly.

B. 500 pounds lighting hoist (2).

C. Platform load capacity of 60 psf with a dead load capacity of 1,000 pounds/6 X 6 inch.

D. Internal lighting system.

PHYSICAL SIZE

Length - 285 inches.
Width - 330 inches.
Height - 416 inches.
Weight - 74,950 pounds.

SUPPLY REQUIREMENTS

120/208 vac, 3 phase, 60 cps, 4 wire.

VOYAGER EVALUATION

This unit should be considered for a design review for use on the Voyager Program. Extensive modifications would be required, however, to facilitate the larger Voyager Spacecraft.

H14-124	SPACECRAFT INTEGRATED SYSTEMS WORKSTAND PART NO. G14-818140 (NAA)

EQUIPMENT SUMMARY DATA SHEET
Sheet 48 of 82

APOLLO FUNCTION SUMMARY

This unit is used for weight and C.G. determination of the S/M and Spacecraft adapter in the Spacecraft checkout area of the MSOB.

APOLLO REQUIREMENTS

A. Unit shall interface with Models A14-154, A14-134, and H14-042.

B. Integrated jacking points and bearing units to provide 2-plane rotation about the base.

C. Will utilize three load cells.

D. Load capacity is not available at present.

VOYAGER EVALUATION

Unit could possibly be used in the Voyager Program but the system load capacity would probably be less than the minimum requirements of the Voyager Spacecraft. Design review would be required to evaluate modifications for increased loading capability of the fixture.

H14-9059	S/M AND SPACECRAFT ADAPTER WEIGHT AND BALANCE FIXTURE PART NO. G17-810003

EQUIPMENT SUMMARY DATA SHEET
Sheet 49 of 82

APOLLO FUNCTION SUMMARY

This unit provides support for the C/M, S/M and spacecraft adapter for determining weight and C.G. measurements in the Spacecraft checkout area of the MSOB.

APOLLO REQUIREMENTS

Load capacity of 30,000 pounds.

PHYSICAL SIZE

Weight - 2,050 pounds.

VOYAGER EVALUATION

This unit should be considered as applicable for use in the Voyager Program. Modification requirements would be only for new jack mating pads to the Spacecraft.

H14-9015-101	WEIGHT AND BALANCE JACK SET PART NO. G16-810012-101 (NAA)

EQUIPMENT SUMMARY DATA SHEET
Sheet 50 of 82

LEM FUNCTION SUMMARY

This unit is used to establish the dry weight and lateral C.G. location of the A/S, D/S and the mated LEM while in the Spacecraft checkout area of the MSOB.

LEM REQUIREMENTS

A. Three load points capable of withstanding a load of 15,000 pounds each with a maximum deflection of 0.0015 inches.

B. Weight determination accuracy within 0.1 percent.

C. C.G. location accuracy within 0.10 inches.

D. Load cells used to measure weight.

SUPPLY REQUIREMENTS

208 vac 3 phase 400 cycle.

PHYSICAL SIZE

Length - 7 feet.

Width - 8 feet.

Height - 5 feet.

Weight - 10,000 pounds.

VOYAGER EVALUATION

This unit has application in the Voyager Program and is capable of supporting the Planetary Vehicle. The mounting points are not compatible with the Voyager vehicle and would require adapter pieces or modification. Weight determination accuracy (0.1 percent) is adequate for Voyager. C.G. location for Voyager is to be determined within ± 0.080 inch in each direction, compared with 0.10 inch accuracy for this item.

WEIGHT AND BALANCE FIXTURE
PART NO. LDW 420-13460 (GAEC)

EQUIPMENT SUMMARY DATA SHEET
Sheet 51 of 82

LEM FUNCTION SUMMARY

This unit provides support of the descent engine during build up operations, especially the injector to thrust chamber connection. It is to be used primarily in the manufacturing areas but is required in the Spacecraft checkout area of the MSOB for descent engine rework.

LEM REQUIREMENTS

A. Mobile structure.

B. Capable of supporting the descent engine.

C. Internal support contured to match the ablative inner lining of the thrust chamber.

PHYSICAL SIZE

Length - 36 inches.

Width - 36 inches.

Height - 33.75 inches.

Weight - 150 pounds.

VOYAGER EVALUATION

If the descent stage engine is selected as the configuration for the Voyager main propulsion, then this fixture would be required for handling capabilities.

DESCENT ENGINE SUPPORT FIXTURE
PART NO. 420-6043 (GAEC)

EQUIPMENT SUMMARY DATA SHEET
Sheet 52 of 82

LEM FUNCTION SUMMARY

This unit provides the capability for installing the flightweight tank assemblies from the underside of the descent stage structure in the Spacecraft checkout area of the MSOB at KSC.

LEM REQUIREMENTS

A. Capable of lifting 90 inches from the floor.
B. Turntable top for alignment.
C. Portable and mobile.
D. Clean room compatible.

PHYSICAL SIZE

Length - 84 inches.
Width - 60 inches.
Height - 90 inches (extended).
Weight - Not available.

VOYAGER EVALUATION

This unit, or a similar one, would have a Voyager application. However, the propulsion system module design may require that modifications be made.

DESCENT STAGE PROPELLANT TANKS INSTALLATION FIXTURE PART NO. LSC-420-63150 (GAEC)

EQUIPMENT SUMMARY DATA SHEET
Sheet 53 of 82

LEM FUNCTION SUMMARY

This unit provides a means of inverting a LEM vehicle at MSC for testing in the mated configuration with the CSM.

LEM REQUIREMENTS

A. Used also in moment of inertia determination test.

B. Can turn the LEM vehicle from its normal attitude to the inverted attitude.

C. Facilities crane used to impart notion.

PHYSICAL SIZE

Length - 20 feet.

Width - 20 feet.

Height - 20 feet (to center of rotation).

Weight - Not available.

VOYAGER EVALUATION

This unit may have application in the Voyager Program for mating tests with the "LANDER" or for moment of inertia tests. It would require loading evaluation and modifications to the vehicle attachment points on the structure.

LEM TURNOVER AND HANDLING FIXTURE
PART NO. LDW 420-13110 (GAEC)

EQUIPMENT SUMMARY DATA SHEET
Sheet 54 of 82

LEM FUNCTION SUMMARY

This unit provides accessibility to both the A/S and D/S when the LEM is mounted on the 3-axis positioner while in the Spacecraft checkout area of the MSOB at KSC.

LEM REQUIREMENTS

A. Capable of supporting the cabin cleanliness enclosure.

B. Three levels of accessibility.

C. Movable on casters.

D. Aluminum construction.

PHYSICAL SIZE

Length - 18 feet.

Width - 18 feet.

Height - 25 feet.

VOYAGER EVALUATION

This unit may be used in the Voyager Program in conjunction with special test fixtures such as the 3-axis positioner. However, the working level heights may require adjustments depending upon final Voyager configuration. Over-all Voyager diameter size would make the unit unacceptable for Voyager use without extensive modifications.

THREE-AXIS POSITIONER WORK PLATFORM
PART NO. 420-13220 (GAEC)

EQUIPMENT SUMMARY DATA SHEET
Sheet 55 of 82

LEM FUNCTION SUMMARY

This unit can accommodate a mated vehicle when it is supported on the 420 - 13700 Support Stand and provide 360 degree access at all levels. It can also support a mated vehicle suspended from the ascent stage sling and system test airspring. The unit divides in half to allow vehicle entry and is used in the Spacecraft checkout area of the MSOB at KSC.

LEM REQUIREMENTS

A. Load capability of 60 psf on the platform surface.

B. Support the mated LEM vehicle from the system test airspring.

C. Provides service hoists on two levels.

SERVICE REQUIREMENTS

A. 115 vac 60 cycle.

B. 90 psi shop air.

PHYSICAL SIZE

Length - 25 feet.

Width - 26 feet.

Height - 28 feet.

Weight - 13 tons.

VOYAGER EVALUATION

The larger Voyager vehicle diameter would require modifications to the work platforms to accommodate the vehicle. Platform heights may also require adjustment depending upon final Voyager configuration. Additional design review would be necessary to determine if this workstand can be used for Voyager checkout.

LEM INTEGRATED WORKSTAND PART NO. 420-13390 (GAEC)

EQUIPMENT SUMMARY DATA SHEET
Sheet 56 of 82

LEM FUNCTION SUMMARY

This unit has the capability to rotate the A/S or D/S about the x, y, or z axes to enable the interior to be vacuumed in the class 1,000,000 clean room at the GAEC, N.Y. facility.

LEM REQUIREMENTS

A. 360 degree rotation at 1 rpm maximum about any 2 axes.

B. Remote control and position indicators.

C. Adapters for mounting either the A/S or D/S vehicles.

SUPPLY REQUIREMENTS

208 vac 3 phase 60 cycle.

PHYSICAL SIZE

Length - 30 feet.

Width - 8 feet.

Height - 13 feet.

Weight - 12,000 pounds.

VOYAGER EVALUATION

This unit would be applicable in some phase of assembly in the Voyager Program. However, its load capacity is smaller than required. Mounting attachment modifications are also necessary. This unit is not acceptable for Voyager use without a complete design review for applicability.

CLEANING POSITIONER PART NO. LDW 420-13750 (GAEC)

EQUIPMENT SUMMARY DATA SHEET
Sheet 57 of 82

LEM FUNCTION SUMMARY

This unit is used for 2-axis rotational checks of the G&N, SCS, RCS, Propulsion and Displays Systems in the KSC MSO Building.

LEM REQUIREMENTS

A. Maximum load of 30,600 pounds.

B. Arc rotation of $\pm$ 6 degrees within $\pm$ 15 minutes.

C. Rotational rates of 4 degrees per second maximum.

D. Vertical axis rotational rate of 1 rpm from 0 to 90 degrees with a 45-degree intermediate stop.

SUPPLY REQUIREMENTS

20 kva, 440 volt, 3 phase 60 cycle.

PHYSICAL SIZE

Length - 12 feet.

Width - 12 feet.

Height - 6 feet.

Weight - 15,000 pounds.

VOYAGER EVALUATION

This unit should be considered for Voyager Planetary Vehicle checkout at the KSC area. The minimum modifications would be the mounting attachment points.

POLARITY CHECKER
PART NO. LSC-420-93089 (GAEC)
G14-818100 (NAA)

EQUIPMENT SUMMARY DATA SHEET
Sheet 58 of 82

LEM FUNCTION SUMMARY

This platform unit is used to provide access to all levels during mated vehicle mounting on the polarity checker. It also provides for mounting of the AOT Optical Target Set during prelaunch checkout tests in the KSC MSO Building.

LEM REQUIREMENTS

A. Provide 180-degree rotation clearance to mated vehicle.

B. Provide 6-degree tilt clearance to mated vehicle.

C. Allow unobstructed line of sight for boresighting of the antenna and the target tower.

D. Accommodate necessary cabling and fluid lines for maintenance and subsystem testing.

SUPPLY REQUIREMENTS

A. 115 vac single phase.

B. 120/208 vac 3 phase 60 cycle.

C. 90 psi shop air.

PHYSICAL SIZE

Length - 297 inches.

Width - 297 inches.

Height - 282 inches.

Weight - 10 tons.

VOYAGER EVALUATION

This unit is applicable for Voyager use with modifications to accommodate the Voyager dimensions and heights.

POLARITY CHECKER WORK PLATFORM PART NO. 420-31040 (GAEC)

EQUIPMENT SUMMARY DATA SHEET
Sheet 59 of 82

LEM FUNCTION SUMMARY

This unit provides a fixture for optically verifying alignment (after A/S and D/S mating) of G&N components, rendezvous radar, S-Band and landing radar antenna, RCS engine thruster axis, and docking and landing reticles. It has provisions for mounting the AOT Optical Target Set and is used in the Spacecraft checkout area of the KSC MSO Building.

LEM REQUIREMENTS

A. Use the LDW 420-13361 Navigation Base Alignment Gage to establish a master coordinate reference frame.

B. Permits checking and corrective action.

VOYAGER EVALUATION

This unit satisfies some basic Voyager requirements for alignment of critical guidance systems but it has very limited flexibility for alignment of anything other than the LEM vehicle. The alignment equipment, mounted on a special Voyager stand, would probably be more practical than extensive fixture modifications.

OPTICAL ALIGNMENT FIXTURE
PART NO. LDW 420-13360 (GAEC)

EQUIPMENT SUMMARY DATA SHEET
Sheet 60 of 82

4.0 SPACECRAFT HANDLING EQUIPMENT

Auxiliary Crane Control, A14-046 (NAA)

Auxiliary Crane Control, A14-134 (NAA)

Auxiliary Crane Control, 420-13060 (GAEC)

Mobile Crane, 420-1290 (GAEC)

Portable Winch Assembly, LDW 420-13311 (GAEC)

S/M Fuel and Oxidizer Tank Sling, H14-060 (NAA)

Spacecraft Sling (without LES), H14-073 (NAA)

Fuel and Oxidizer Tank Sling, H14-102 (NAA)

Spacecraft Sling, CSM, SLA, LEM, H14-212 (NAA)

Descent Engine Installation Sling, 420-63500 (GAEC)

Descent Engine Turnover Sling, LSC 420-63511 (GAEC)

LEM Inverted Hoisting Sling, LDW 420-13068 (GAEC)

LEM Vehicle Hoisting Sling, 420-1300 (GAEC)

Battery Handling Mobile Hoist Rig, 420-83109 (GAEC)

Descent Stage Handling Dolly, 420-13550 (GAEC)

Descent Stage Propellant Tank Dolly, LDW 420-63980 (GAEC)

Descent Stage Engine Installation Dolly, 420-63400 (GAEC)

SIVB LEM Adapter Base Assembly, H14-165-101 (NAA)

SIVB LEM Adapter Base Assembly, H14-166-101 (NAA)

APOLLO FUNCTION SUMMARY

This unit provides hoisting operations with precision linear movements and weight and strain indications. It can interface with all general purpose cranes and can be used in all test areas.

APOLLO REQUIREMENTS

A. Load capacity of 40,000 pounds.

B. Capable of raising or lowering a load 12 inches with an accuracy of 0.001 inches.

C. Lower speed of 0 to 8 feet per minute maximum.

D. Remote control.

SUPPLY REQUIREMENTS

None

PHYSICAL SIZE

Length - 12 inches.
Width - 12 inches.
Height - 34 inches (retracted).
Weight - 750 pounds.

VOYAGER EVALUATION

This unit would be applicable for service at manufacturing, assembly, and field checkout mating facilities on the Voyager Program. No modifications would be necessary, as the unit is greater than the 30,000-pound required load capacity of the Voyager Space Vehicle.

A14-046 AUXILIARY CRANE CONTROL
PART NO. G14-820001 (NAA)

EQUIPMENT SUMMARY DATA SHEET
Sheet 61 of 82

APOLLO FUNCTION SUMMARY

This unit provides hoisting operations with precision linear movement and weight and strain indications. It can interface with all general-purpose cranes and can be used in all test areas.

APOLLO REQUIREMENTS

A. Load capacity of 20,000 pounds.

B. Lowering rate of 0 to 8 feet per minute.

C. Lower a distance of 12 inches to an accuracy of 0.001 inches maximum.

D. Safety factor of 5.0.

SUPPLY REQUIREMENTS

100 psig GN_2.

PHYSICAL SIZE

Length - 9 inches.

Width - 9 inches.

Height - 43 inches (retracted).

Weight - 225 pounds.

VOYAGER EVALUATION

This unit can be used without modifications in the Voyager Program in manufacturing, assembly, and checkout areas, but would be limited to specific applications with loads of less than 20,000 pounds.

A14-134	AUXILIARY CRANE CONTROL PART NO. G14-820020 (NAA)

EQUIPMENT SUMMARY DATA SHEET
Sheet 62 of 82

LEM FUNCTION SUMMARY

This unit provides a control unit for minute positioning of the ascent or descent stage, mated vehicle or other LEM components during handling operations at any test facility. It will interface with all general-handling crane equipment.

LEM REQUIREMENTS

A. Self-contained, hydraulically operated interconnecting unit between the lifting device and the appropriate hoisting sling.

B. Load capacity of 10 tons.

C. Lowering distance of 12 inches with an accuracy within 0.001 inch.

D. 40-foot remote operations capability.

E. Built-in load gauge.

SUPPLY REQUIREMENTS

100 psig maximum air pressure.

PHYSICAL SIZE

Length - 9 inches.
Width - 9 inches.
Height - 45.25 inches fully extended.
Weight - 208 pounds.

VOYAGER EVALUATION

This unit is similar to the NAA A14-134 crane control. Such a unit is applicable to the Voyager Program but does not have the necessary load capacity to handle the complete Voyager bus.

AUXILIARY CRANE CONTROL
PART NO. 420-13060 (GAEC)

EQUIPMENT SUMMARY DATA SHEET
Sheet 63 of 82

LEM FUNCTION SUMMARY

This unit is a general purpose, vehicle-mounted crane used in the manufacturing and assembly area.

LEM REQUIREMENTS

A. Load capacity of 20 tons.

B. 50-foot boom length.

C. Gasoline operated.

D. Power-lowered hoist drum instead of gravity system.

PHYSICAL SIZE

Length - 96 inches.

Width - 300 inches.

Height - 141 inches.

Weight - 25 tons.

VOYAGER EVALUATION

This unit would have application for special "boilerplate" type vehicle test and for general handling, loading or unloading, on the Voyager Program. Identical units may be available from military surplus.

MOBILE CRANE
PART NO. 420-1290 (GAEC)

EQUIPMENT SUMMARY DATA SHEET
Sheet 64 of 82

LEM FUNCTION SUMMARY

This unit is used for hauling a loaded cargo pallet into the Pregnant Guppy aircraft at the manufacturing area by connecting to existing aircraft tie points.

LEM REQUIREMENTS

A. Portable electric winch.

B. Maximum line pull of 6,000 pounds at 10 fpm.

C. 17-foot usable cable length.

SUPPLY REQUIREMENTS

A. 120 volt, 3 phase, 400 cycle ac.

B. Built-in adapter for use with diesel generator.

PHYSICAL SIZE

Length - 24 inches.

Width - 24 inches.

Height - 12 inches.

Weight - 52 pounds

VOYAGER EVALUATION

This unit must be considered for use in any Voyager vehicle transportation by the Pregnant Guppy (B-377 PG aircraft). Transportation requirements are at present undefined, but this unit can be used for pulling a pallet or skid into the plane and would be compatible with the B-377 PG aircraft.

PORTABLE WINCH ASSEMBLY
PART NO. LDW 420-13311 (GAEC)

EQUIPMENT SUMMARY DATA SHEET
Sheet 65 of 82

APOLLO FUNCTION SUMMARY

This unit facilitates the handling of the fuel and oxidizer (main propellant) tanks during installation and removal from the service module, tank support, and shipping containers. The unit is used in all manufacturing and assembly areas, and in the Spacecraft checkout area of the MSOB.

APOLLO REQUIREMENTS

A. Load capacity 760 pounds (ultimate load of 2,280 pounds).

B. Provides shielding for the tank domes and a S/M guard ring to prevent S/M damage during installation of the tanks.

C. Swivel 360 degrees about the vertical axis.

PHYSICAL SIZE

Weight - 74 pounds.

VOYAGER EVALUATION

The unit should be considered as an accessory item to be used in the propulsion systems assembly in Valley Forge or in field service at the launch site. Attachment point modifications will be necessary.

H14-060	FUEL AND OXIDIZER TANK SLING PART NO. G17-818031 (NAA)

EQUIPMENT SUMMARY DATA SHEET
Sheet ~~66~~ of 82

APOLLO FUNCTION SUMMARY

This unit provides the handling and transporting capability of a stacked C/M, S/M, S-IVB adapter, and support base, but without the LES. It is used in the VAB and the Spacecraft Checkout area of the MSOB.

APOLLO REQUIREMENTS

A. Load capacity of 32,000 pounds.

B. Allows a 5-inch radial C.G. shift for C.G. alignment.

C. Use in level "E" clean room.

D. Interfaces with No. 15 Vulcon Hook and three LES tower attachment points on the C/M

E. Used for general handling requirements with the Spacecraft in the vertical position.

PHYSICAL SIZE

Weight - 1,080 pounds.

VOYAGER EVALUATION

The unit should be considered for use in the handling of the Planetary Vehicle with modifications at the attachment points. The unit would require modifications to cover the spacecraft spread for lifting points, however, and could impose overload conditions to the present configuration of the sling.

H14-073 SPACECRAFT SLING (WITHOUT LES)
PART NO. G14-818002 (NAA)

EQUIPMENT SUMMARY DATA SHEET
Sheet 67 of 82

APOLLO FUNCTION SUMMARY

This unit is to be used in the handling of fuel, oxidizer, and pressurization tanks for installation into or removal from the service module in the Spacecraft Checkout area of the MSOB.

APOLLO REQUIREMENTS

A. Load capacity of 5,000 pounds.

B. Interface with a 15-ton crane hook and with the tank attachment points.

C. Use in level "E" clean room.

D. Two legs with adjustable turnbuckles.

VOYAGER EVALUATION

The load capacity of this unit is greater than for the H14-060 unit and it is more of a general usage-type piece of GSE. It should be considered for use in the Voyager Program in much the same manner. It would require adapters for the attachment points.

H14-102	FUEL AND OXIDIZER TANK SLING PART NO. G17-818232 (NAA)

EQUIPMENT SUMMARY DATA SHEET
Sheet 68 of 82

APOLLO FUNCTION SUMMARY

This unit will be used to hoist and handle the Apollo spacecraft command service module (CSM), SLA, LEM, and associated stacked configuration GSE in the Spacecraft Checkout area of the MSOB or in the VAB.

APOLLO REQUIREMENTS

A. Attaches to the four LES attach points.

B. Interfaces with a 25 or 50 ton hyro-set.

C. Load capacity of 65,000 pounds at a maximum C.G. excursion of 4 inches.

D. Proof load of 2.7 times design load.

PHYSICAL SIZE

Weight - 3,000 pounds.

VOYAGER EVALUATION

This unit meets the required Voyager load capacities but modifications would have to be made to the attachment points of the sling to the Voyager spacecraft. Unit has rigid frame with eye-bolt adjustable cables with special end fittings for command module attachment.

H14-212 SPACECRAFT SLING, CSM, SLA, LEM PART NO. 8N14-810060 (NAA)

EQUIPMENT SUMMARY DATA SHEET
Sheet 69 of 82

LEM FUNCTION SUMMARY

This sling is used to support the suspended weight of the STL descent stage rocket engine assembly in a vertical attitude during removal and installation operations. The sling is used in conjunction with a suitable lifting device or crane in manufacturing operations and at the Spacecraft Checkout facilities in the KSC MSO Building.

LEM REQUIREMENTS

A. Rated load at 500 pounds and proof loaded at 1,000 pounds.

B. Compatible with STL descent stage engine.

PHYSICAL SIZE

Length - 23 inches.

Width - 20 inches.

Height - 30 inches.

Weight - 30 pounds.

VOYAGER EVALUATION

This unit must be considered for Voyager Program application if the STL descent engine is part of the Voyager configuration. No modifications would be required to the unit.

DESCENT ENGINE INSTALLATION SLING **PART NO. 420-63500 (GAEC)**

EQUIPMENT SUMMARY DATA SHEET
Sheet 70 of 82

LEM FUNCTION SUMMARY

This unit supports the descent assembly in a horizontal attitude. It also provides the capability for manually rotating the engine 90 degrees, while suspended, to enable the engine to be removed from the shipping container and lowered onto the descent engine dolly. It is used at both the manufacturing facility and the KSC Spacecraft Checkout area in the MSOB.

LEM REQUIREMENTS

A. Rated load of 500 pounds.
B. Proof loaded at 2.67 times rated load.
C. Manual 90 degrees rotation about the "y" axis.

PHYSICAL SIZE

Length - 44 inches.
Width - 2.5 inches.
Height - 42 inches.
Weight - 100 pounds.

VOYAGER EVALUATION

If the (LEM) descent engine assembly is selected for Voyager, this unit should be applicable without modifications.

DESCENT ENGINE TURNOVER SLING PART NO. LSC 420-63511 (GAEC)

EQUIPMENT SUMMARY DATA SHEET
Sheet 71 of 82

LEM FUNCTION SUMMARY

This sling is used in handling an inverted LEM vehicle for mating tests with the CSM during docked configuration thermal vacuum tests at MSC.

LEM REQUIREMENTS

A. Load weight support of 30,000 pounds.

B. Provides leveling capability for CSM mating.

C. Mates with overhead hoists.

PHYSICAL SIZE

"X" frame I-beam 228 inches square.

VOYAGER EVALUATION

This sling can be used in the Voyager Program, if it is required, as a general handling piece of equipment; however, it would require modifications to the four cable attachment points for the vehicle interface.

LEM INVERTED HOISTING SLING
PART NO. LDW 420-13068 (GAEC)

EQUIPMENT SUMMARY DATA SHEET
Sheet 72 of 82

LEM FUNCTION SUMMARY

This sling provides support capability for a fully ballasted mated LEM vehicle during descent vibration testing at the Bethpage, Long Island manufacturing site.

LEM REQUIREMENTS

A. Support vehicle weight of 32,500 pounds.

B. Proof loaded to 87,000 pounds.

C. Use with low frequency suspension system and the Apex fitting adapters.

PHYSICAL SIZE

Length - 232 inches.

Width - 232 inches.

Height - 198 inches.

Weight - 1,400 pounds.

VOYAGER EVALUATION

This sling can be used in the Voyager Program with only modifications to the vehicle interface connecting points.

LEM VEHICLE HOISTING SLING
PART NO. 420-1300 (GAEC)

EQUIPMENT SUMMARY DATA SHEET
Sheet 73 of 82

LEM FUNCTION SUMMARY

This rig provides a method of handling the primary batteries when no handling facility exists in both the Spacecraft Checkout areas of the MSOB and VAB as well as the MSS at the launch pad.

LEM REQUIREMENTS

A. Lifting height of 84 inches.

B. Lift battery weighing 140 pounds.

C. Hoist and move battery to desired location.

PHYSICAL SIZE

Length - 60 inches.
Width - 32 inches.
Height - 96 inches.
Weight - 200 pounds.

VOYAGER EVALUATION

This unit is a general purpose type battery handling mobile rig and could be used in the Voyager Program. Considerations must be given for the battery supporting sling compatibility with Voyager battery configuration.

BATTERY HANDLING MOBILE HOIST RIG
PART NO. 420-83109 (GAEC)

EQUIPMENT SUMMARY DATA SHEET
Sheet 74 of 82

LEM FUNCTION SUMMARY

This unit (or special configurations of it) is designed to handle the descent stage on the aircraft pallet, the cold flow room, and the EMI testing room.

LEM REQUIREMENTS

A. Support a maximum dynamic load of 12,000 pounds and static load of 22,000 pounds when jack supported.

B. Towing capability at 8 mph maximum.

C. Compatible with B-377 PG aircraft.

PHYSICAL SIZE

Length - 160 inches (mounting points).

Width - 160 inches (mounting points).

Height - 49 inches (mounting points).

VOYAGER EVALUATION

The load capacities of this unit would have to be increased for use in the Voyager Program. Also, an adapter plate may be necessary to compromise the mounting point incompatibility. Design of a new unit with Voyager requirements would probably be more effective.

DESCENT STAGE HANDLING DOLLY
PART NO. 420-13550 (GAEC)

EQUIPMENT SUMMARY DATA SHEET
Sheet 75 of 82

LEM FUNCTION SUMMARY

This unit is used to support and transport a descent stage propellant tank without inducing unfavorable loads into the tank structure. It is used in the Spacecraft checkout area of the MSOB.

LEM REQUIREMENTS

A. Support a 200-pound stage tank.

B. Must be compatible with clean room requirements.

C. Provide damage and environment protection.

PHYSICAL SIZE

Length - 68.5 inches.
Width - 63 inches.
Height - 88 inches (including cover).
Weight - 600 pounds.

VOYAGER EVALUATION

This unit could be used for the Voyager Program if the descent stage engine and propellant tank assembly are used as the main propulsion system for Voyager. It could be modified for other tank configuration mounting points but the external unit cover would limit the overall Voyager tank dimensions that would be handled by the unit.

DESCENT STAGE PROPELLANT TANK DOLLY
PART NO. LDW 420-63980 (GAEC)

EQUIPMENT SUMMARY DATA SHEET
Sheet 76 of 82

LEM FUNCTION SUMMARY

This unit provides support and positioning capability for the descent stage engine when installing it in the descent stage structure. It is used in the manufacturing areas as well as the Spacecraft checkout area of the MSOB at KSC.

LEM REQUIREMENTS

A. Rotational capability from vertical to horizontal and back with a hand crank.

B. Provides engine mobility during handling.

PHYSICAL SIZE

Length - 72 inches.

Width - 103 inches.

Height - 90 inches.

Weight - 720 pounds.

VOYAGER EVALUATION

This unit would be applicable without modifications for handling the descent stage engine for the Voyager Program. However, modifications may be required for adequate space vehicle clearances during the installation and removal of the engine.

DESCENT STAGE ENGINE INSTALLATION DOLLY
PART NO. 420-63400 (GAEC)

EQUIPMENT SUMMARY DATA SHEET
Sheet 77 of 82

APOLLO FUNCTION SUMMARY

This unit provides protection for the aft interface of the S-IVB adapter, and supports the adapter, with the LEM/CSM mated, during transportation, storage, maintenance, and checkout in the KSC test facilities.

APOLLO REQUIREMENTS

A. Load capacity of 41,000 pounds maximum.

B. Interface with the A14-157 LEM adapter base closure, H14-173 dolly and leveling jack set H14-162.

PHYSICAL SIZE

Weight - 4,000 pounds

Inside Diameter - 245 inches.

Outside Diameter - 275 inches.

Height - 37.25 inches.

VOYAGER EVALUATION

This unit would be compatible with any presently planned S-IVB adapter package on the aft side and therefore should be considered for the Voyager Program. Modifications may be required for mounting points to the Voyager Spacecraft, but only if the Voyager - S-IVB-IU interface is of a different configuration than the LEM-S-IVB interface.

H14-165-101	S-IVB - LEM ADAPTER BASE ASSEMBLY PART NO. G24-818040 (NAA)

EQUIPMENT SUMMARY DATA SHEET
Sheet 78 of 82

APOLLO FUNCTION SUMMARY

This unit will support and protect the upper interface of the spacecraft LEM adapter (SLA) with and without the Spacecraft during transportation, handling, and checkout within the KSC test facilities.

APOLLO REQUIREMENTS

The requirements are the same as for the H14-165 unit except that they will support a SLA/Spacecraft with the S/M tanks containing 37,400 pounds of deionized water.

PHYSICAL SIZE

Inside Diameter - 220 inches.

Outside Diameter - 240 inches.

Height - 30 inches.

Weight - 4,000 pounds.

VOYAGER EVALUATION

This unit would not be applicable for use in the Voyager Program. The unit would satisfy the LEM-CSM interface requirement which is not required on Voyager.

H14-166-101 S-IVB-LEM ADAPTER BASE ASSEMBLY
PART NO. G24-818110 (NAA)

EQUIPMENT SUMMARY DATA SHEET
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5.0 TRANSPORTING EQUIPMENT

Spacecraft Vertical Transport Vehicle, H14-131 (NAA)

Base Closure, S-IVB LEM Adapter, A14-157 (NAA)

Wheeled Warehouse Tractor, Gas Powered, 420-13330 (GAEC)

APOLLO FUNCTION SUMMARY

This unit will transport a C/M-S/M plus support base in a vertical position at 5 mph within the KSC test facilities.

APOLLO REQUIREMENTS

A. Load capacity of 30,000 pounds.
B. Maximum speed of 5 mph.
C. Tires 11.00 X 20 - tandem axle.
D. Braking system.
E. Compatible with M-48 tractor or commercial equivalent.

PHYSICAL SIZE

Load Area - 160" X 180".

Trailer size - 160" Width X 400" Length.

Weight -

VOYAGER EVALUATION

This unit could be used in the Voyager Program with a special spacecraft support base to handle the Voyager vehicle. This could be a permanent modification to the trailer or a GSE adapter unit.

H14-131	SPACECRAFT VERTICAL TRANSPORT VEHICLE PART NO. ME 183-0024-0001 (NAA)

EQUIPMENT SUMMARY DATA SHEET
Sheet 80 of 82

APOLLO FUNCTION SUMMARY

This unit provides environmental protection for the SLA from contamination through the base opening, during on-site storage and transporation within all test areas.

APOLLO REQUIREMENTS

Fabric will be waterproof (vinyl) coated nylon fabric.

PHYSICAL SIZE

A. Covers an opening of 245 inches diameter.

B. Weight - 50 pounds.

VOYAGER EVALUATION

This unit would be considered for the Voyager Program but due to limited life characteristics of the unit material, it probably would not be serviceable when needed. Modifications would be necessary, however, for attachment hardware and straps.

A14-157	BASE CLOSURE, S-IVB (LEM) ADAPTER PART NO. G24-828100 (NAA)

EQUIPMENT SUMMARY DATA SHEET
Sheet 81 of 82

LEM FUNCTION SUMMARY

This unit is a general purpose self-propelled towing vehicle used in moving wheeled assemblies or units from area to area within the KSC facility.

LEM REQUIREMENTS

A. Draw bar pull of 4,000 pounds at 14 mph.

B. Gasoline powered engine.

C. Turning radius - 10 feet (outside).

PHYSICAL SIZE

Length - 64 inches.

Width - 114 inches.

Height - 57 inches.

Weight - 5,600 pounds.

VOYAGER EVALUATION

This unit or a similar one would be applicable for general purpose transporting of wheeled equipment. Identical units may be available in military surplus capacity.

WHEELED WAREHOUSE TRACTOR, GAS POWERED PART NO. 420-13330 (GAEC)

EQUIPMENT SUMMARY DATA SHEET
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